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THE INFLUENCE OF SHOCK WAVE-BOUNDARY LAYER EFFECTS ON THE DESIGN OF HYPERSONIC AIRCRAFT

RICHARD D. NEUMANN

GERALD L. BURKE

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FOREWORD

This report was prepared by the Gasdynamics Branch, Flight Mechanics Division, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio. The work was accomplished under Project No. 1366, "Aerodynamics and Flight Mechanics," Task No. 13C607, "Hypersonic Gasdynamic Heating." The report was written by Richard D. Neumann (FDMG), Technical Manager for Aerothermodynamics, and Gerald L. Burke (FDMG), Aerospace Engineer. This report covers correlations conducted between January 1967 and July 1967 on experimental data taken under USAF Contract No. AF33(615)-1202. Portions of this report were presented at the AIAA Guidance, Control and Flight Dynamics Conference in Huntsville, Alabama 14-16 August 1967. The report was then revised and submitted in October 1968.

This technical report has been reviewed and is approved.

Philip P. Antonatos

PHILIP P. ANTONATOS
Chief, Flight Mechanics Division
Air Force Flight Dynamics
Laboratory

ABSTRACT

The design of aircraft for sustained operation at hypersonic speeds requires the understanding of aerodynamic heating generated through interfering flow fields. Such interactions not only determine the required level of vehicle thermal protection but also create severe gradients of temperature along skin panels. The Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, has completed an extensive experimental program supporting the conceptual design of these vehicles in which experimental results have been generated on models illuminating the basic features of both two- and three-dimensional interactions with results applicable to the design of hypersonic aircraft. This report presents these data and correlations with theory in the Mach number range 6 through 10. Results indicate the applicability of current design practices, areas requiring further investigation, and the problems involved in interpretation and application of interference data from hypersonic facilities to the desired free flight condition.

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ILLUSTRATIONS

ILLUSTRATIONS (Cont)

LIST OF SYMBOLS

A	proportionality constant: 0.332 for laminar flow; 0.0296 for turbulent flow
C	constant in linear viscosity law
C_p	specific heat at constant pressure (assumed constant)
h	heat transfer coefficient (BTU/ft^2-sec-°R)
K_3	pressure gradient term in heat transfer theory from Reference 8
M	Mach number
n	exponent: 0.5 for laminar flow; 0.2 for turbulent flow
P	absolute pressure
Re	Reynolds number per foot
St	Stanton number
T	absolute temperature
U	flow velocity
X	length on model on free stream direction (ft unless otherwise denoted)
δ	deflection angle of shock generator
θ	fin shock angle (see Figure 16)
Λ	leading edge sweep angle of fin
ϕ	ray angle of line of peak heating (see Figure 16)

Subscripts

e	inviscid value
FIN_{LE}	length based on location of fin leading edge
MAX	maximum value in the impingement region
P_{MAX}	Based on maximum pressure conditions
REF	reference condition based on laminar or turbulent flow

LIST OF SYMBOLS (Cont)

TURB	based on assumed turbulent flow
$*$	evaluated at Eckert's reference temperature
Λ	evaluated at a particular sweep angle
∞	free stream condition

SECTION I

INTRODUCTION

The design of aircraft to attain and sustain flight at hypersonic speeds requires accurate determination of the induced thermal environment experienced during flight. The configurational complexities of such aircraft, caused by the requirement for compatibility of both high and low speed performance and the mating of airframe and power plant, create regions of both severe and localized heating. These regions must be adequately evaluated and minimized in order to assure structural efficiency and system practicality. This localized aerodynamic heating and its prediction at the flight Mach numbers for first generation cruise systems is presented in this report.

Two- and three-dimensional interaction data are presented over a range of test conditions. Results of earlier authors, notably the work of Sayano (Reference 1), and Fabish and Levin (Reference 2) on the two-dimensional problem and the work of Miller, et al. (Reference 3) and Stainback (Reference 4) on the three-dimensional interactions are extended to the higher Mach numbers and unified to present a more complete picture of these basic interactions. Correlations of the data are presented which allow the design engineer to rapidly estimate the peak interference heating. In the case of the three-dimensional fin interaction, an improved design method is presented which extends the work of Miller and Redeker (Reference 5) by relaxing the dependence of the method on the levels and gradients of measured pressure data.

SECTION II

TEST PROGRAM

The experimental program was conducted in the 50-inch hypersonic facilities of the Arnold Engineering Development Center (AEDC) at Mach numbers 6 and 8 in Tunnel B and at Mach 10 in Tunnel C. A nominally laminar test point was run at a unit Reynolds number of 10^6/ft while a nominally turbulent test point was run at 3.5×10^6/ft.

A highly instrumented sharp plate was used to measure the characteristic interaction created by both two- and three-dimensional shock generators. The two-dimensional generators were remote planar surfaces of variable incidence with respect to the flow along the flat plate. The three-dimensional generators were fins of variable sweep angle and incidence. The test configurations are shown in Figures 1 and 2. Measurements on the flat plate included both surface static pressure and heat transfer distributions. Separate plate inserts were used for each measurement to assure complete data coverage at each spatial location.

The two-dimensional generator was slightly blunted to create a stronger interaction and to separate the generator shock from the nonuniform generator trailing edge expansion process creating a longer region on the plate in which to observe the effects of impingement.

The three-dimensional generators were sharp fins with sweep angles of 0, 45, 60, and 75 degrees. Local fin incidence was varied from 5 to 20 degrees with the majority of data taken at 7.5 and 15.0 degrees.

Turbulent boundary layers on the plate were assured through the use of a trip device shown in Figure 3. Static and total pressure measurements taken on the plate and in the plate flow field indicated the existence of fully developed turbulent flow which was free of trip induced distortions in interaction regions where the data were obtained.

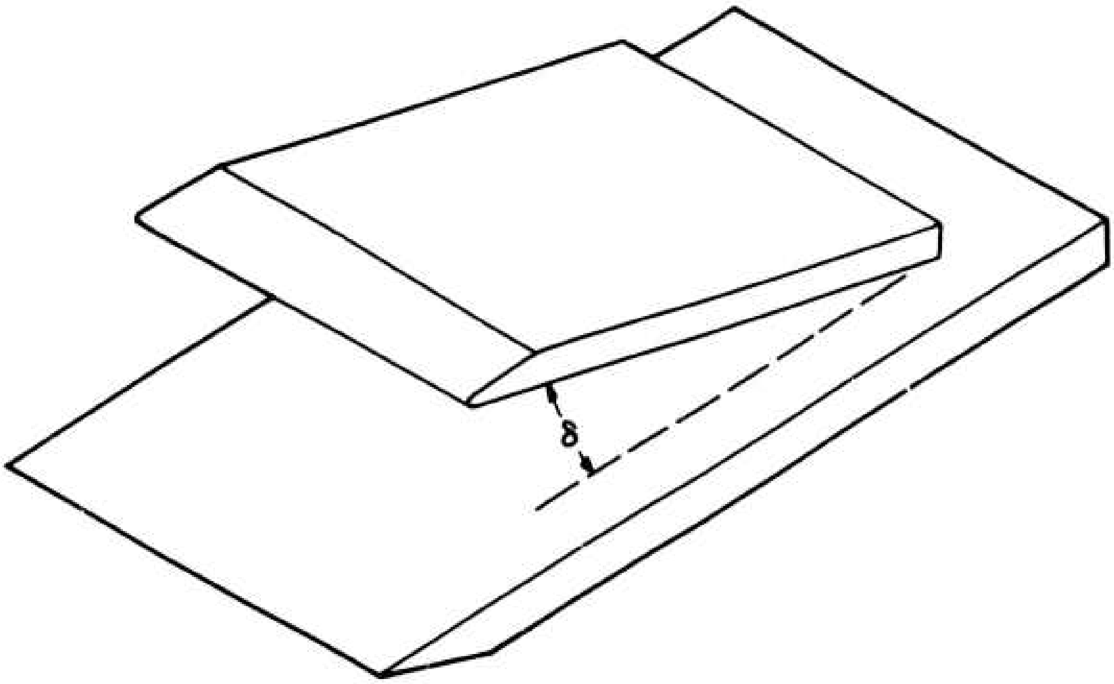


Figure 1. Two-Dimensional Interaction Model

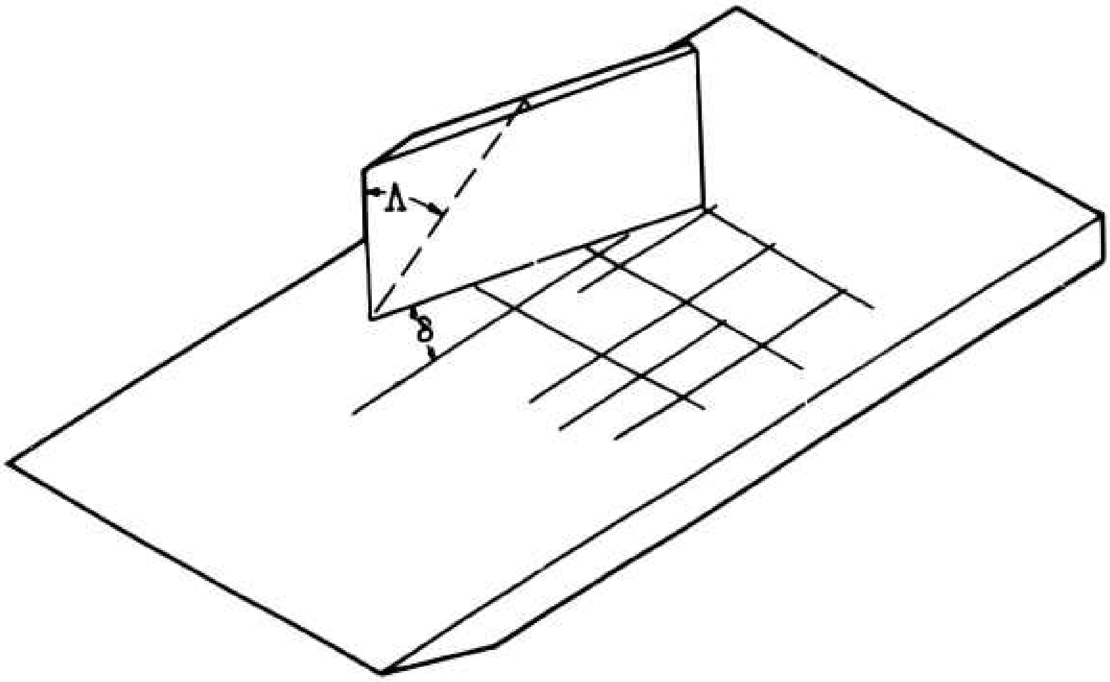


Figure 2. Three-Dimensional Interaction Model

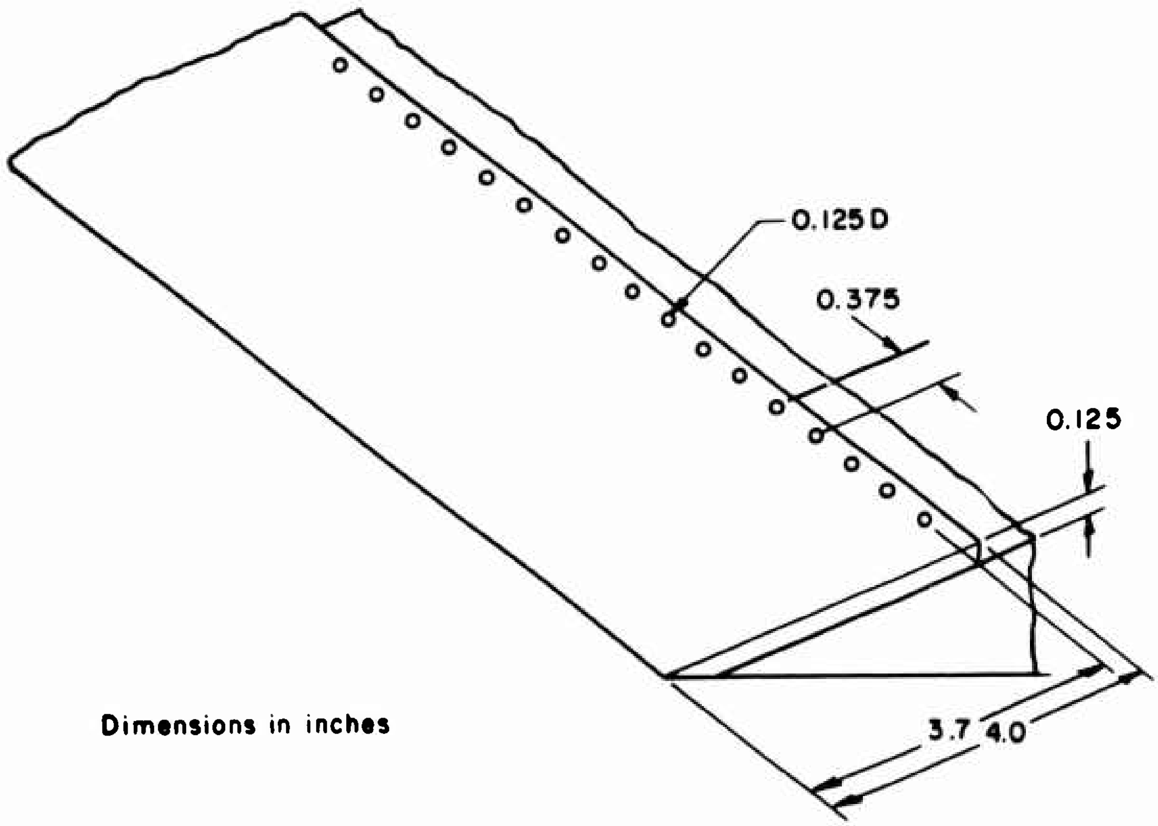


Figure 3. Trip Device on Turbulent Flat Plate Model

SECTION III

DISCUSSION OF CORRELATIONS

1. BASIC FLAT PLATE DATA

Data were taken on the flat plate model alone to verify flow quality in the interaction region and to establish reference heat transfer coefficients against which to evaluate the interaction data. Two test points were employed: the nominally laminar test point at a unit Reynolds number of 10^6/ft and the nominally turbulent test point at 3.5 x 10^6/ft.

a. Laminar Test Point

Laminar reference heat transfer data were evaluated as follows:

M_∞	$Re_\infty / ft \times 10^{-6}$	$hX^{0.5} \times 10^3$
6	0.999	1.23
8	0.980	0.97

where

$$\left[hX^{0.5} \right] = \left[BTU - in.^{0.5} / ft^2 - sec - {}^\circ R \right]$$

Corresponding turbulent reference flat plate data were not measured at this condition but were analytically estimated by relating the laminar and turbulent heating through the reference temperature method. The relationship, derived in Appendix I, is stated as

$$\frac{h_{TURB}}{h_{LAM}} = 0.0892 \, (Re_{\infty_x})^{0.3} \left(\frac{T^*}{T_\infty}\right)^{0.6}$$

This relationship assumes turbulent flow to have initiated at the plate leading edge and is therefore suspect to the extent of that assumption. However, its application in this report is limited to an understanding of interactions near the plate leading edge and since turbulent heating varies only as the 0.2 power of distance, the method is considered acceptable for the present application.

b. Turbulent Test Point

Reference heat transfer data for the case of an artificially induced turbulent boundary layer were measured and are presented as follows:

M_∞	$Re_\infty/ft \times 10^{-6}$	$hx^{0.2} \times 10^3$
6	3.46	3.05
8	3.35	2.45

where

$$\left[hx^{0.2} \right] \left[\frac{BTU - in.^{0.2}}{ft^2 - sec - {}^\circ R} \right]$$

Laminar reference heat transfer data for this test point were evaluated through use of the previously described laminar data recalling that $S_T(Re_\infty)^{1/2} =$ constant for constant angle-of-attack flow. The resulting reference data are as follows:

M_∞	$Re_\infty \times 10^{-6}$	$hx^{0.5} \times 10^3$
6	3.46	2.29
8	3.35	1.79

Corresponding pressure data were obtained to determine surface static pressure gradients. The laminar static pressures are shown in Figure 4 and the turbulent data in Figure 5. An adverse pressure gradient is noted in the turbulent data. This is due to the presence of the trip mechanism in the leading edge region which produces a locally separated flow. An adverse pressure gradient upon reattachment disturbs the flow field upstream of the interaction region. The areas on the plate where both two- and three-dimensional impingements were observed are shown in these figures. The figures also show that small pressure gradients existed in the interaction region but that the local pressure was very nearly equal to the free stream static value. Because of the small errors introduced, it is assumed in the following analysis that free stream pressure existed locally on the plate in the interaction region.

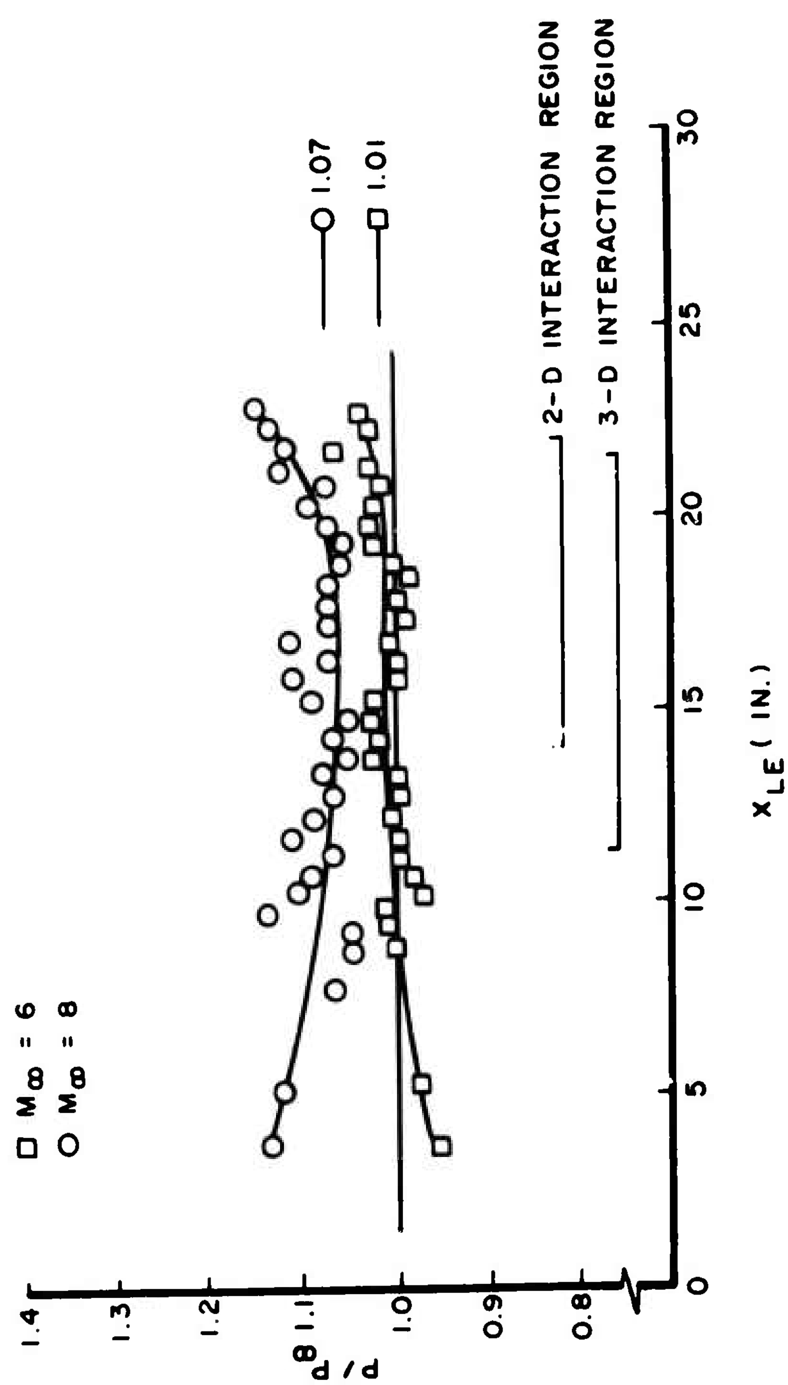


Figure 4. Laminar Static Pressure Distribution on the Plate

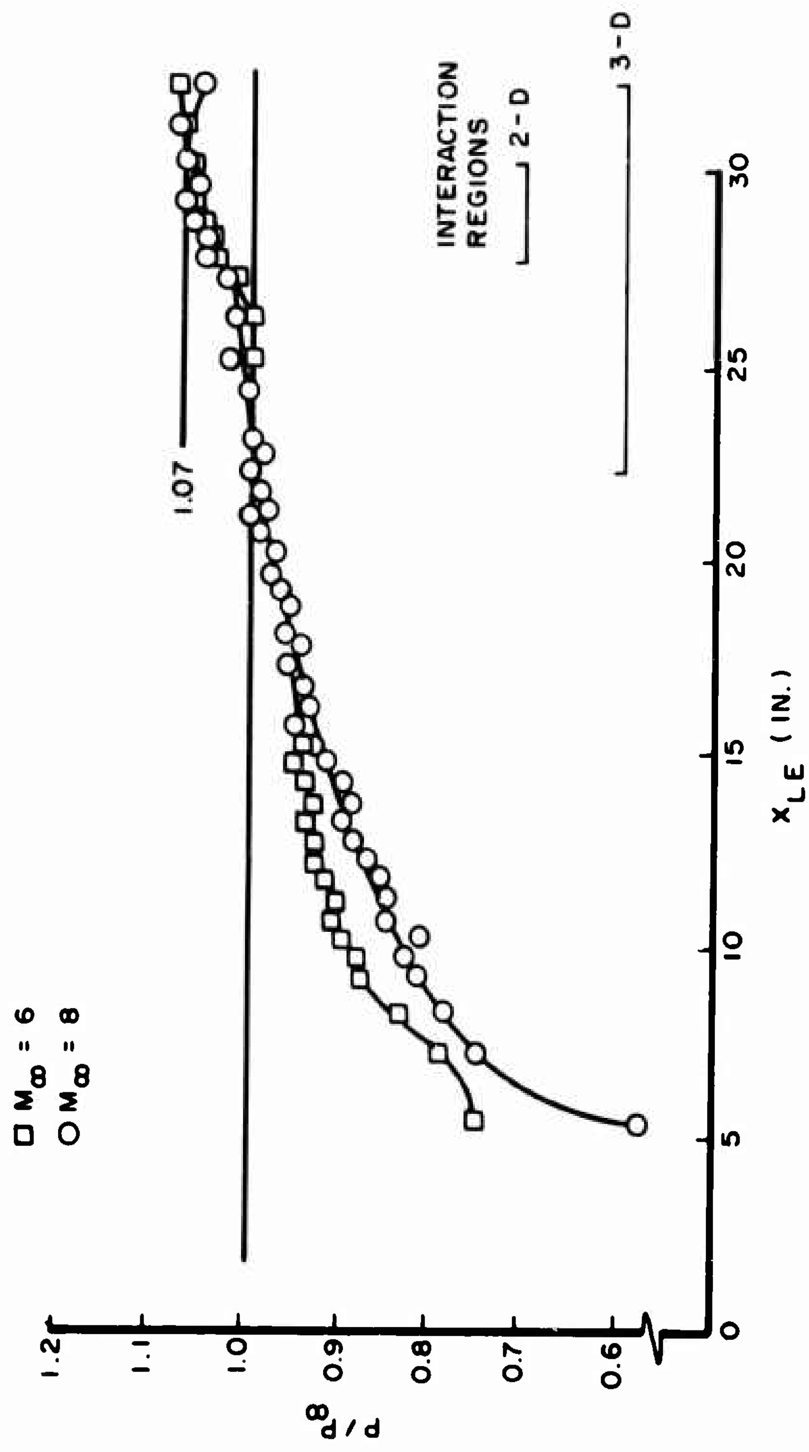


Figure 5. Turbulent Static Pressure Distribution on the Plate

2. TWO-DIMENSIONAL INTERACTIONS

The first experimental model evaluated exhibited a two-dimensional interaction. (See Figure 1 for the model orientation.) This model configuration is essentially that of References 1 and 2 and the program objectives were to generate both additional, more detailed data at higher shock strengths, and laminar data in the interaction region.

Both pressure and heat transfer rates were measured from which the classic relationship between interference heat transfer and pressures known as the pressure interaction theory was developed (see Appendix II for theory development). Figure 6 indicates the turbulent data acquired during this test program at Mach 6, 8, and 10 as well as supporting data of Sayano (Reference 1) and Fabish and Levin (Reference 2). Excellent correlation is noted for all data. Laminar data for the same configuration are plotted in Figure 7. Shown also on this graph are supporting data from Holden (Reference 6) and Kutschenreuter (Reference 7). A distinct lack of agreement with theory is shown in this figure. Data slopes, for most cases, vary with the power of the pressure ratio as derived from turbulent theory. The data are further segmented on separate but parallel lines with seemingly arbitrary separation.

Transition was suspected as the cause of this disagreement and the data were ratioed to the turbulent reference value using the analytic expression derived in Appendix I.

$$\frac{h_{TURB}}{h_{LAM}} = 0.0892 \ (Re_{\infty_x})^{0.3} \left(\frac{T^*}{T_\infty}\right)^{0.6}$$

Using this reference value, the correlation was significantly improved. In Figure 8 the data at Mach 6 and 8 as well as the high Reynolds number data of Kutschenreuter (Reference 7) are observed to correlate about the turbulent theory line. Two sets of data, our data at Mach 10 and data presented by Holden (Reference 6) at nearly the same Mach number, deviate from the theory at low deflection angles but agree with turbulent theory at the higher shock strength, indicating transitional behavior.

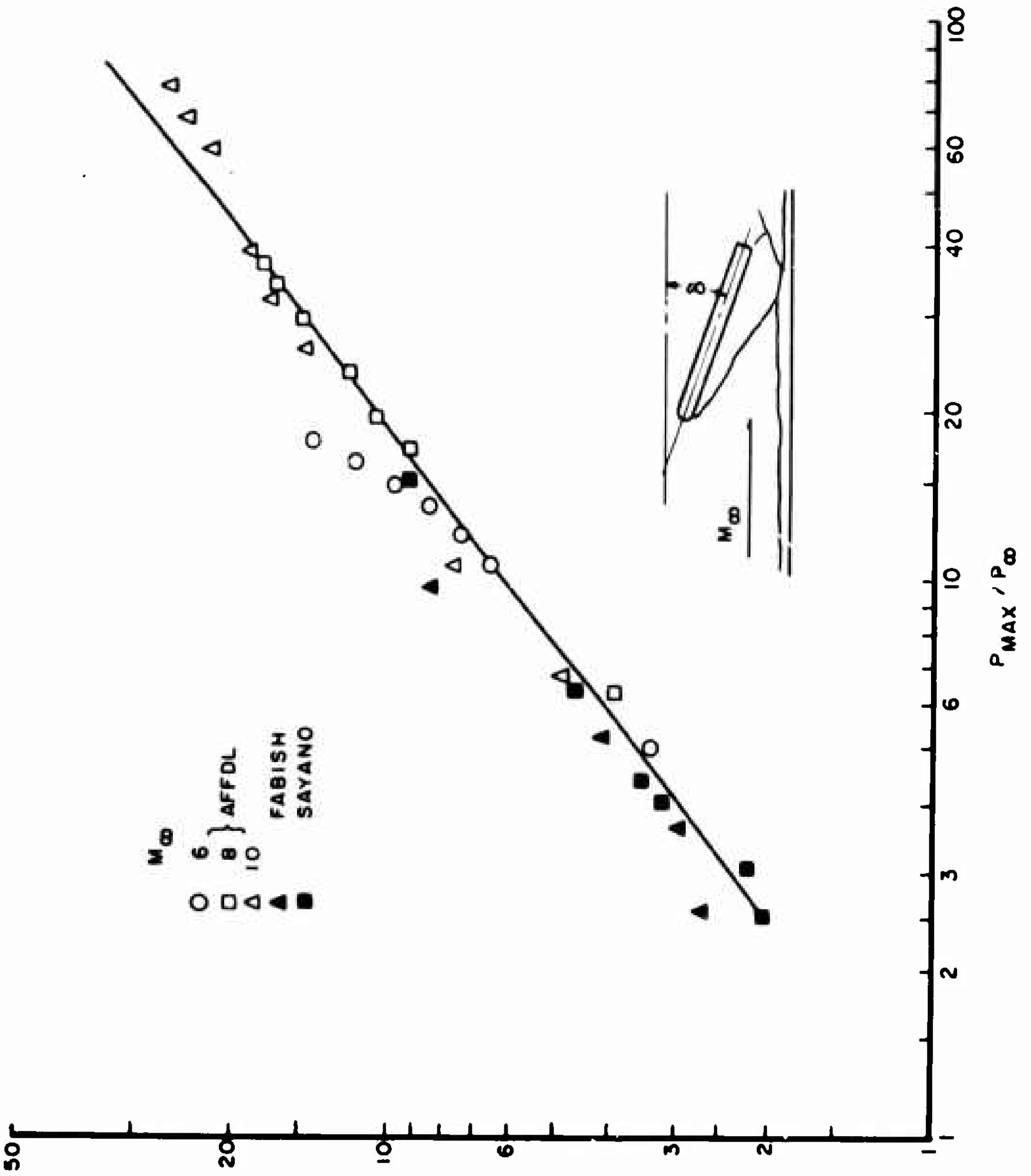


Figure 6. Turbulent Two-Dimensional Interaction Data

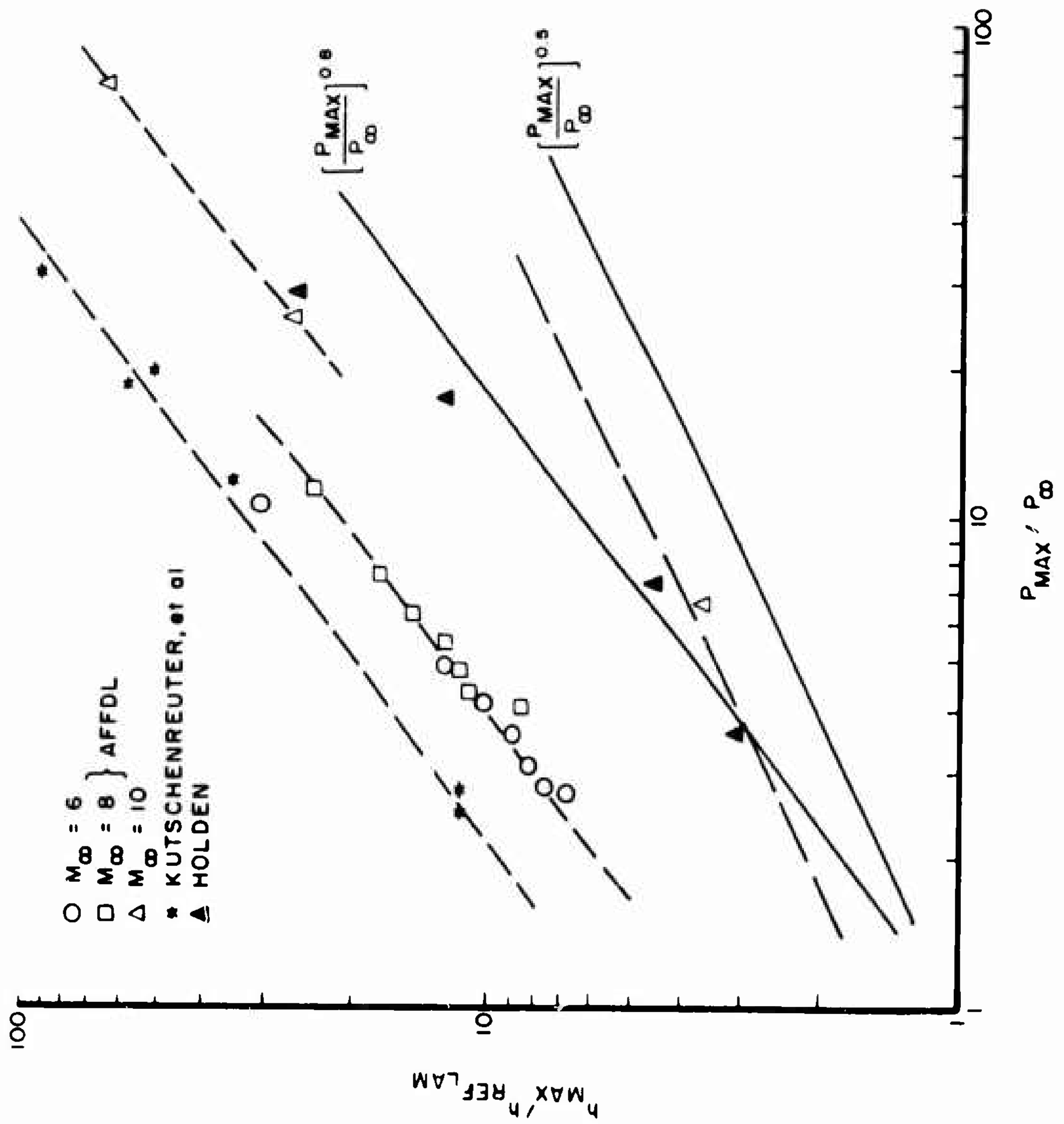


Figure 7. Laminar Two-Dimensional Interaction Data

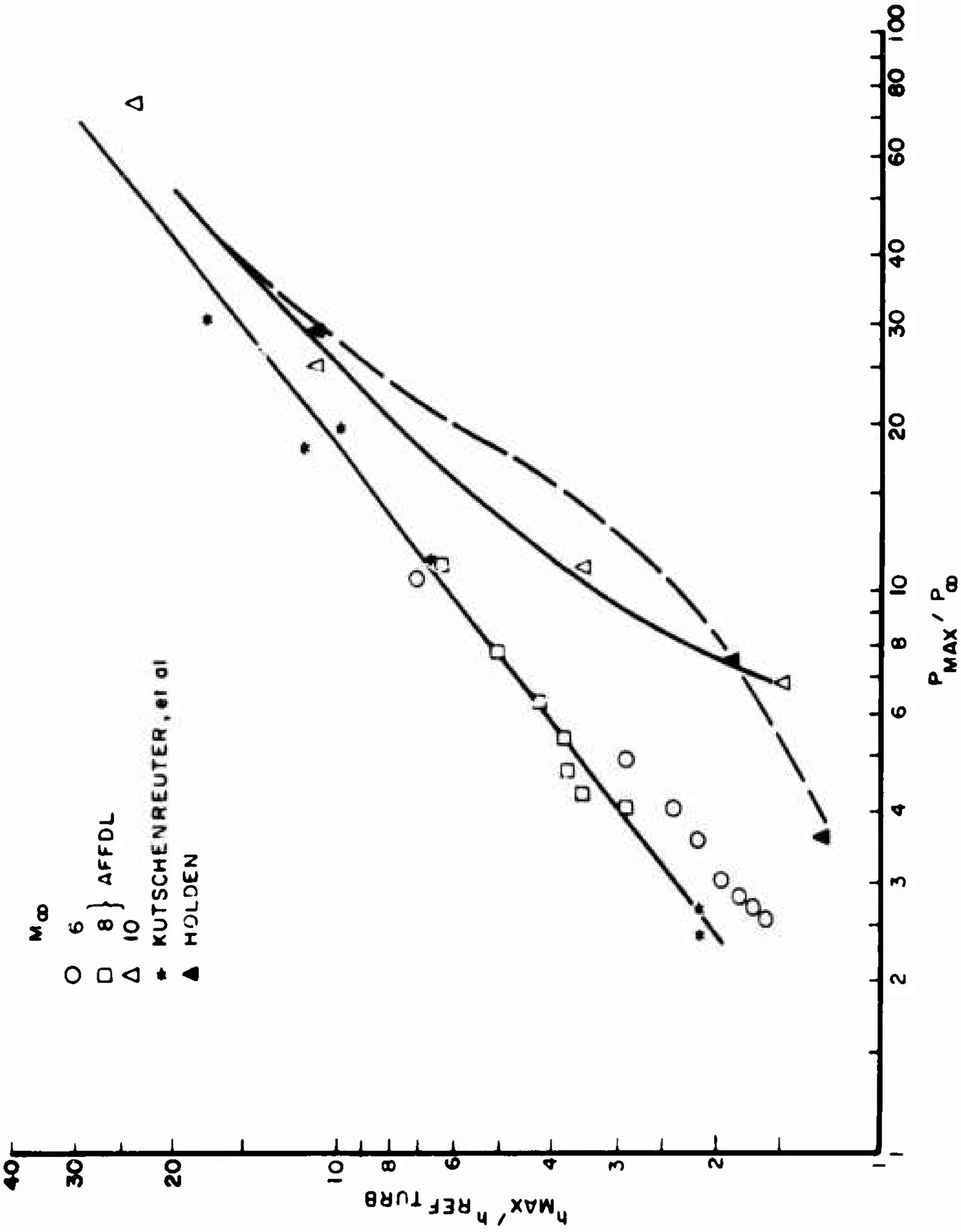


Figure 8. Initially Laminar Two-Dimensional Interaction Data

The present series of tests as well as data from several diverse sources have indicated the acceptability of the pressure interaction theory and in predicting such interaction heating. Further, the apparent tendency of the boundary layer to become turbulent in the interaction region indicates the prudence of using the turbulent correlation for design studies in the low hypersonic Mach number regime.

3. THREE-DIMENSIONAL INTERACTIONS

The second experimental model which was evaluated was the three-dimensional model depicted in Figure 2.

a. Interference Pressures

Pressures in the fin impingement region on the instrumented plate were evaluated against oblique shock theory. For the case of no fin sweep, the results of the correlation were excellent. Measured peak pressures in the interaction region showed no effect of the separation process ahead of the fin shock system or of the boundary layer state in the impingement region and agreed with oblique shock theory within 7%. Due to this agreement and the fact that complementary pressure data were not available for each heat transfer run, oblique shock theory was used in the later correlations of the heat transfer data.

The data with fin sweep proved more difficult and was correlated in a slightly different manner. It was noted that the pressure in the interaction region did not decay significantly with sweep. Correlation of the swept fin data was achieved through the use of exponential pressure decay with the cosine of the sweep angle as shown in Figure 9. The solid symbols indicate the values obtained from the oblique shock relations used at zero fin sweep and the open symbols show data for various sweep and local incidence angles. The straight lines are fairings of the data. The resultant empirical expression for peak impingement pressure becomes

$$\left. \frac{P_{\Lambda}}{P_{\Lambda=0}} \right|_{\delta = \text{const}} = \cos^{0.3} \Lambda$$

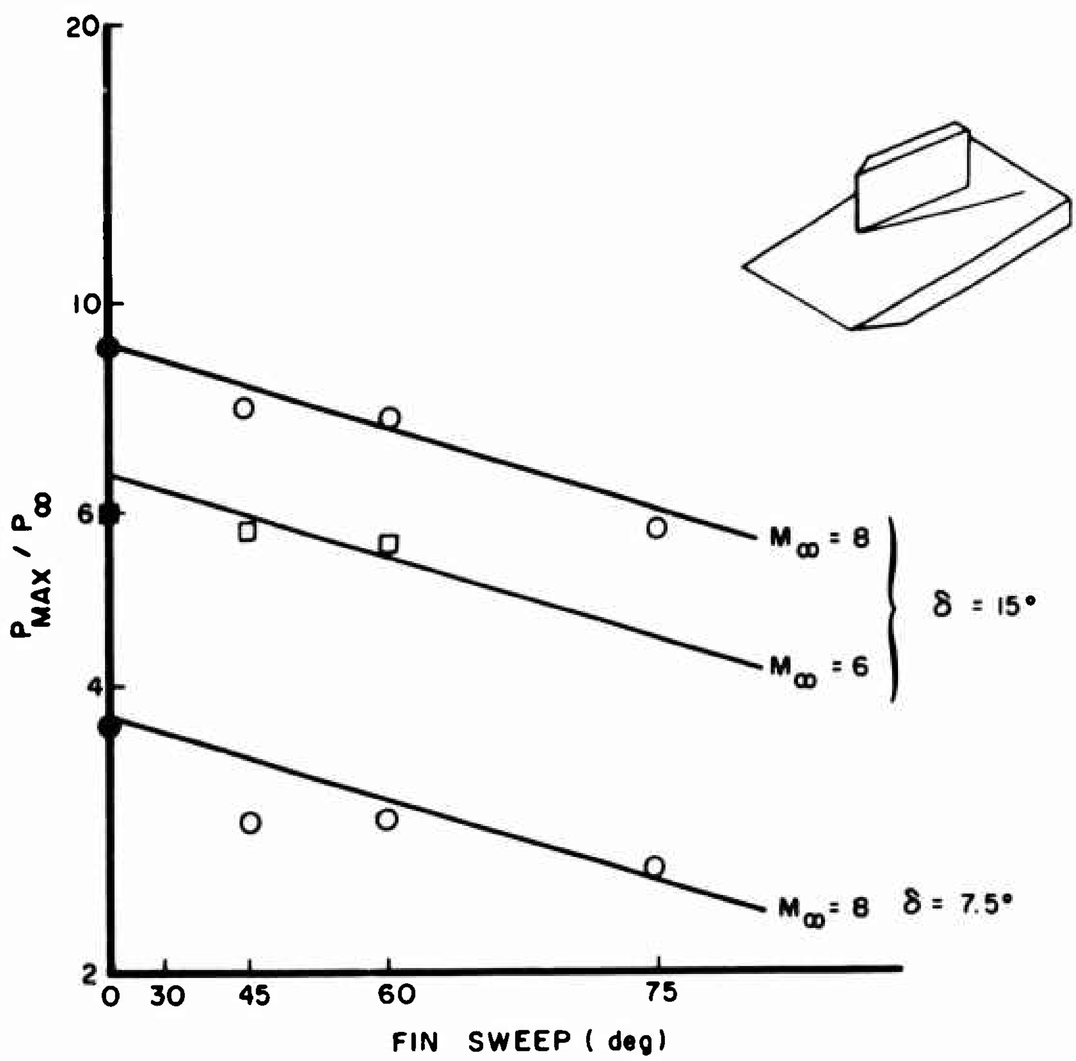


Figure 9. Effect of Sweep on the Maximum Pressure in the Fin Interference Region

A more general view of the peak pressure data may be obtained through the use of the hypersonic similarity relationship and the sweep dependence relation previously discussed. Figure 10 indicates all the data taken in the interaction region and the overall agreement of the data with theory. It can be seen that interaction pressure ratios between 2 and 11 are generated at Mach numbers of 6 and 8. Although higher interaction strengths are indicated at Mach 10, the lack of adequate heat transfer instrumentation prohibited a meaningful evaluation of these data.

b. Interference Heat Transfer Data

(1) Laminar Boundary Layer Case

Let us consider first the heat transfer data generated by a sharp and unswept fin in an initially laminar boundary layer on the plate. Following the work of Miller, the pressure interaction theory was used to correlate the data. Unlike the two-dimensional interaction case previously considered, several chordwise rows of heat transfer gages were intersected by the fin shock system allowing local heat transfer maxima to be measured and distributions of maxima over the plate to be analyzed.

In the initial observation of the fin interaction data, the magnitude of heating rates for the laminar boundary layer case were of a magnitude as to suggest that transition had occurred. This was considered to be analogous to the two-dimensional case discussed previously. To verify this, the date were correlated against the turbulent form of the pressure interaction theory. This correlation for the Mach 6 data is shown in Figure 11b and for the Mach 8 data in Figure 12b.* These correlations indicate that, although the turbulent theory values are numerically equivalent to the data, the trend with distance (as shown by the progression of symbols) does not agree with theory. Similar

*The progression of symbols in these figures from the circle through the diamond indicates data taken along the line of peak heating with increasing distance from the fin leading edge. The gages represented are approximately five inches apart in the streamwise direction with the most forward gage 4.5 inches downstream of the fin leading edge.

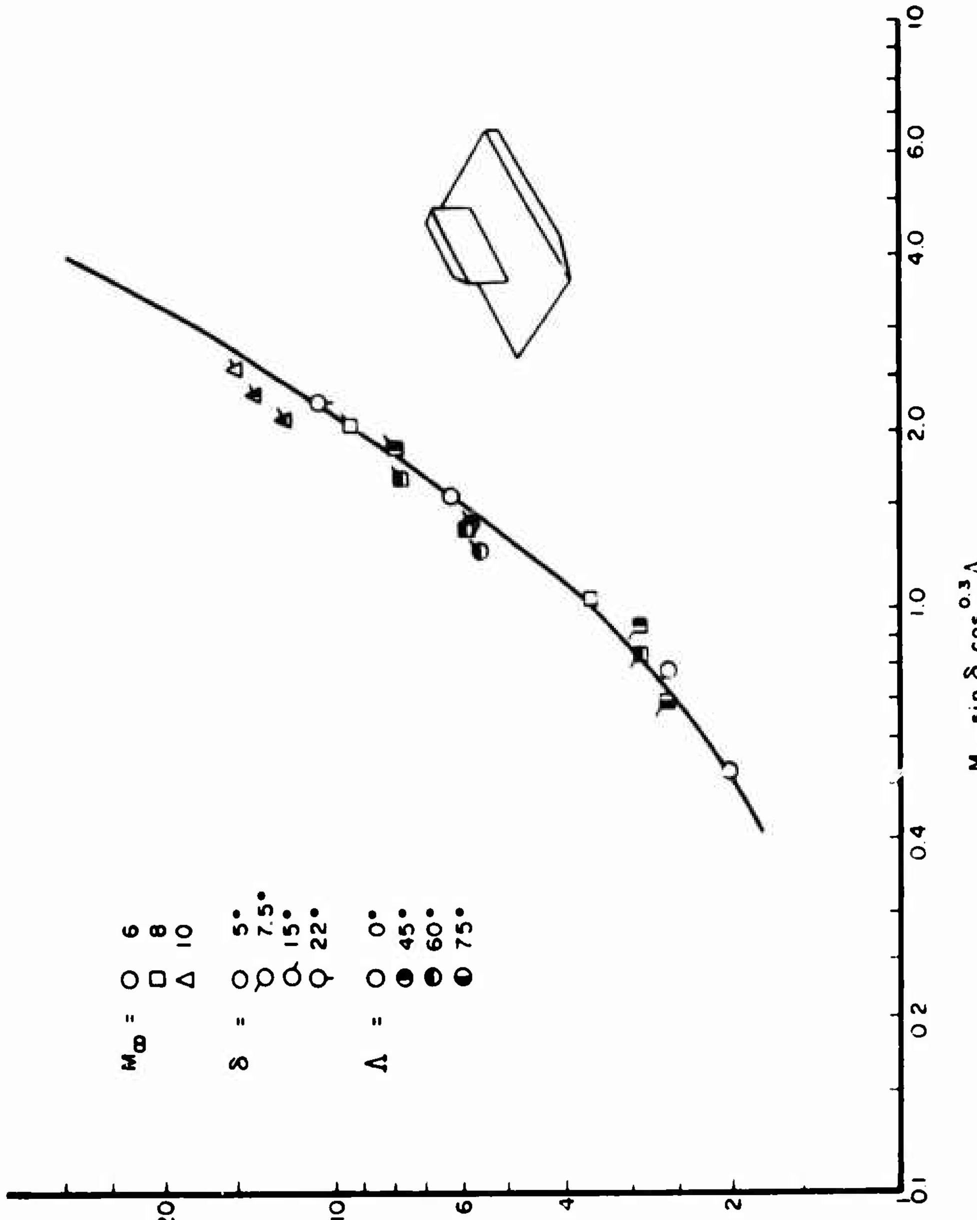


Figure 10. Correlation of Peak Pressure Data Measured in the Fin Interference Region

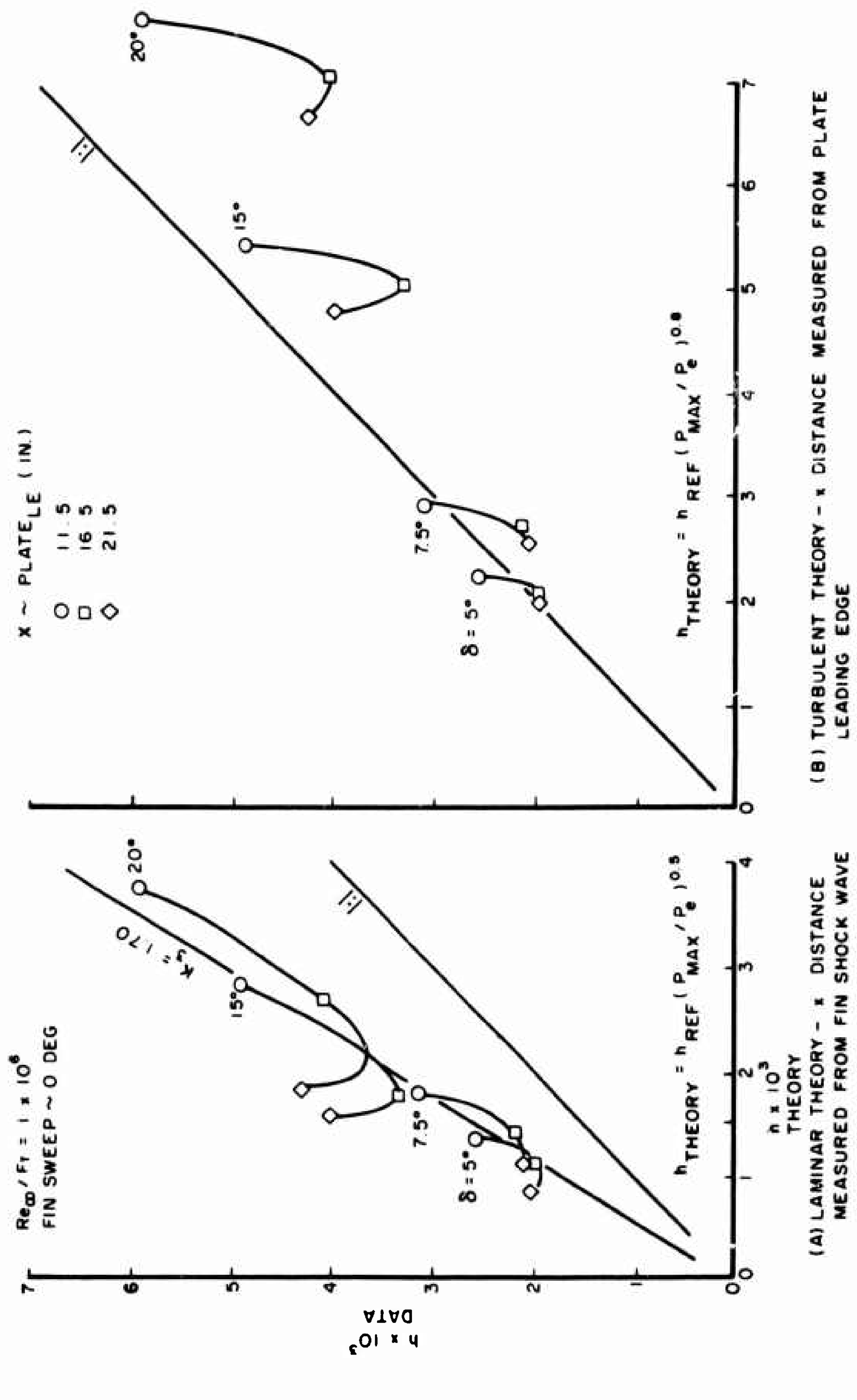


Figure 11. Correlation of Mach 6 Laminar Fin Interaction Data

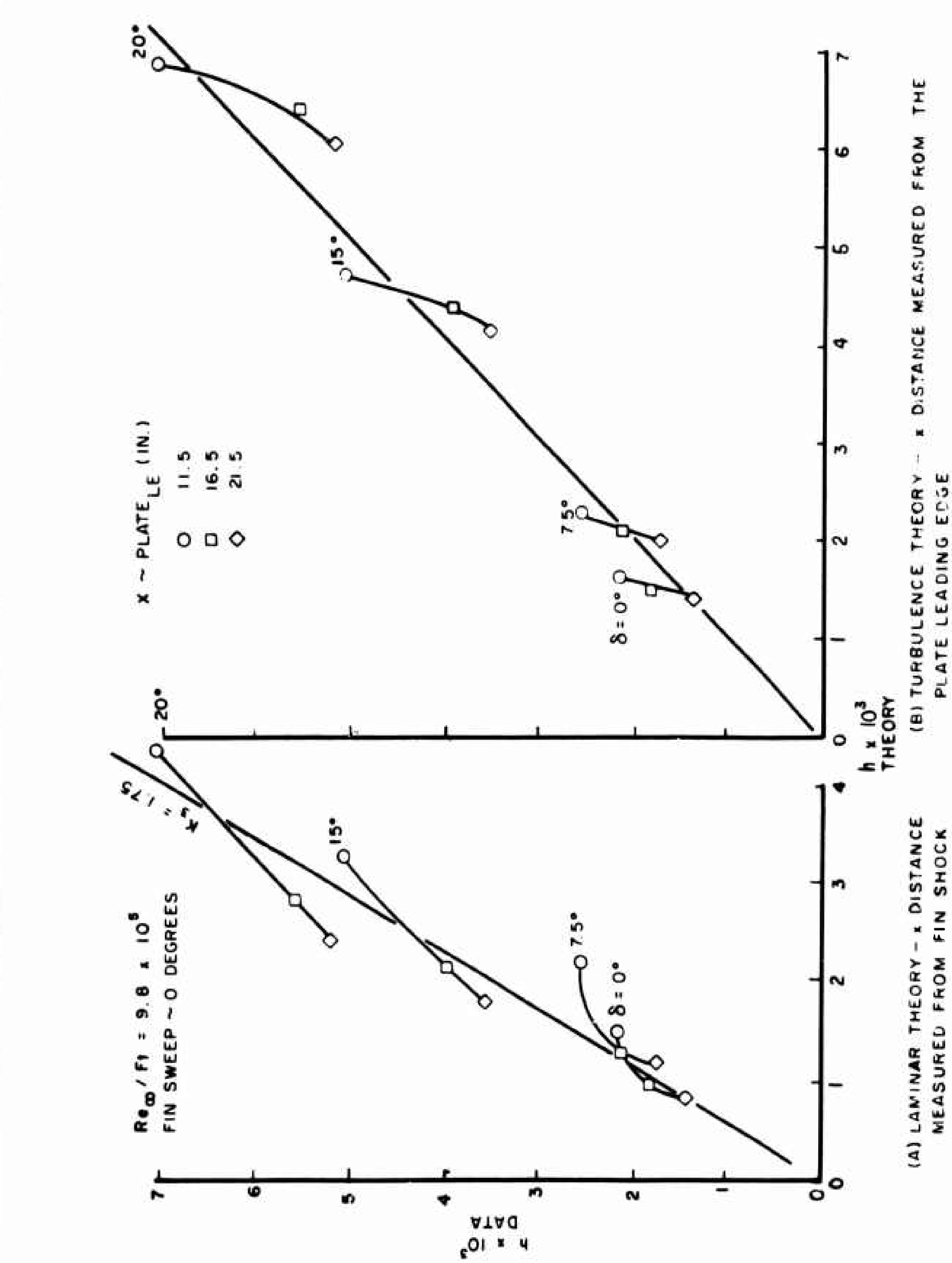


Figure 12. Correlation of Mach 8 Laminar Fin Interaction Data

correlation using the laminar form of the interaction theory was also attempted as shown in Figure 13. While the ratio of data to the laminar theory was high (almost a factor of 5 neglecting the pressure gradient term in the theory), the correlation trends were significantly improved. Assuming a constant value of K_3 (pressure gradient term in the heat transfer theory from Reference 8) indicative of very large adverse pressure gradients (K_3 of the order of 2), the data are still severely underpredicted by the laminar theory by a factor of 2.5.

Reviewing the variables in the pressure interaction theory and following a suggestion of Hankey (Reference 9) that the three-dimensional impingement process initiates a new boundary layer, a reference length for the flat plate heat transfer coefficient was postulated. This reference length was defined as the distance between the inviscid fin shock location (from oblique shock theory) and the peak heat transfer gage location. Further, it was assumed that the flow direction in this region was parallel to the free stream velocity vector. Using this application of the pressure interaction theory, data were again plotted against a form of the laminar theory (the product of the heat transfer coefficient evaluated at this new reference length and the square root of the oblique shock pressure rise). As shown in Figures 11a and 12a, the slope of the line correlating data with theory reduced to a factor of 1.70 for Mach 6 data and 1.75 for Mach 8 data. This factor was assumed to be the correction to the similar solution approach accounting for adverse pressure gradients. These values were considered reasonable in light of work of Bertram and Feller (Reference 8), although in a more exacting analysis, slight variations as a function of shock strength should be expected and were, in fact, detected in correlation of the data. Using this method, distributions of peak heat transfer in the interaction region at both Mach 6 and Mach 8 were calculated and compared with data as shown in Figures 14 and 15.

In order to generalize the proposed method and detach it from reliance on test data, the location of each heat transfer maximum was empirically determined. It was found that most data lie along a ray angle, ϕ , which has a unique relation with the inviscid shock angle θ in the Mach 6 to 8 data range. The relation derived was that $\phi = 0.785\theta$. This correlation, shown in Figure 16, was found to be insensitive to the boundary layer state although the turbulent

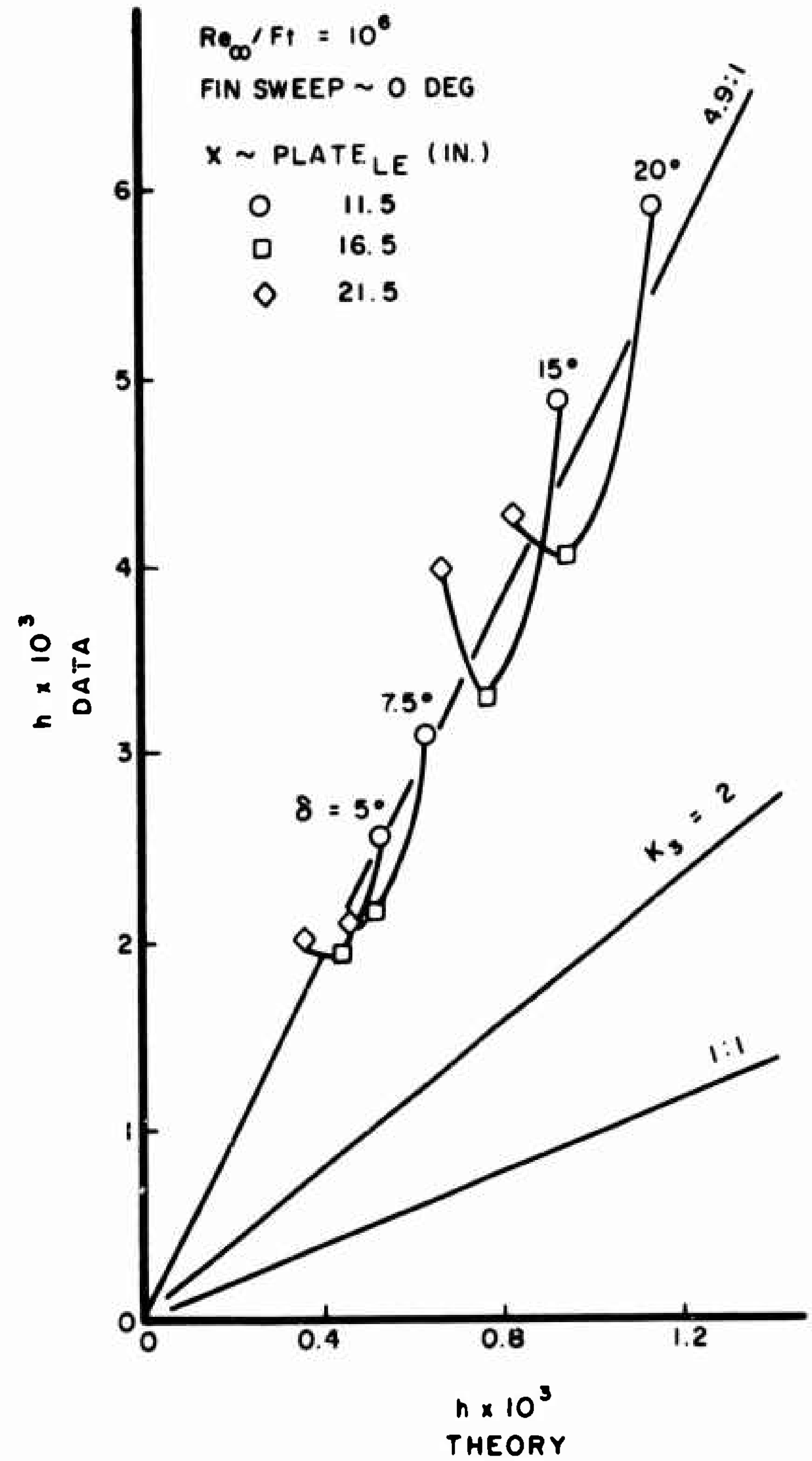


Figure 13. Correlation of Mach 8 Fin Interaction Data With Laminar Theory

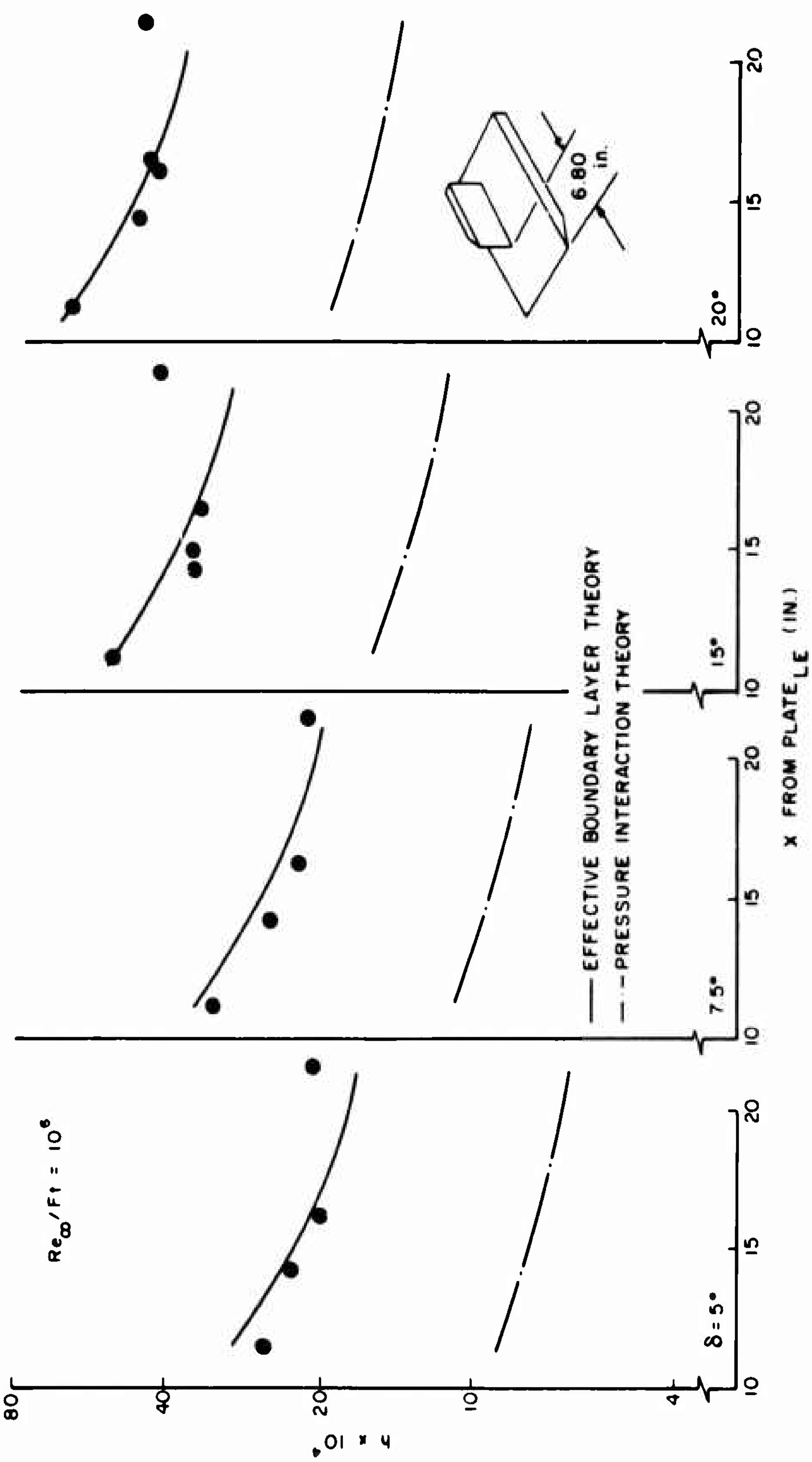


Figure 14. Correlation of the Maximum Heating in the Fin Interaction Region - Mach 6 Laminar

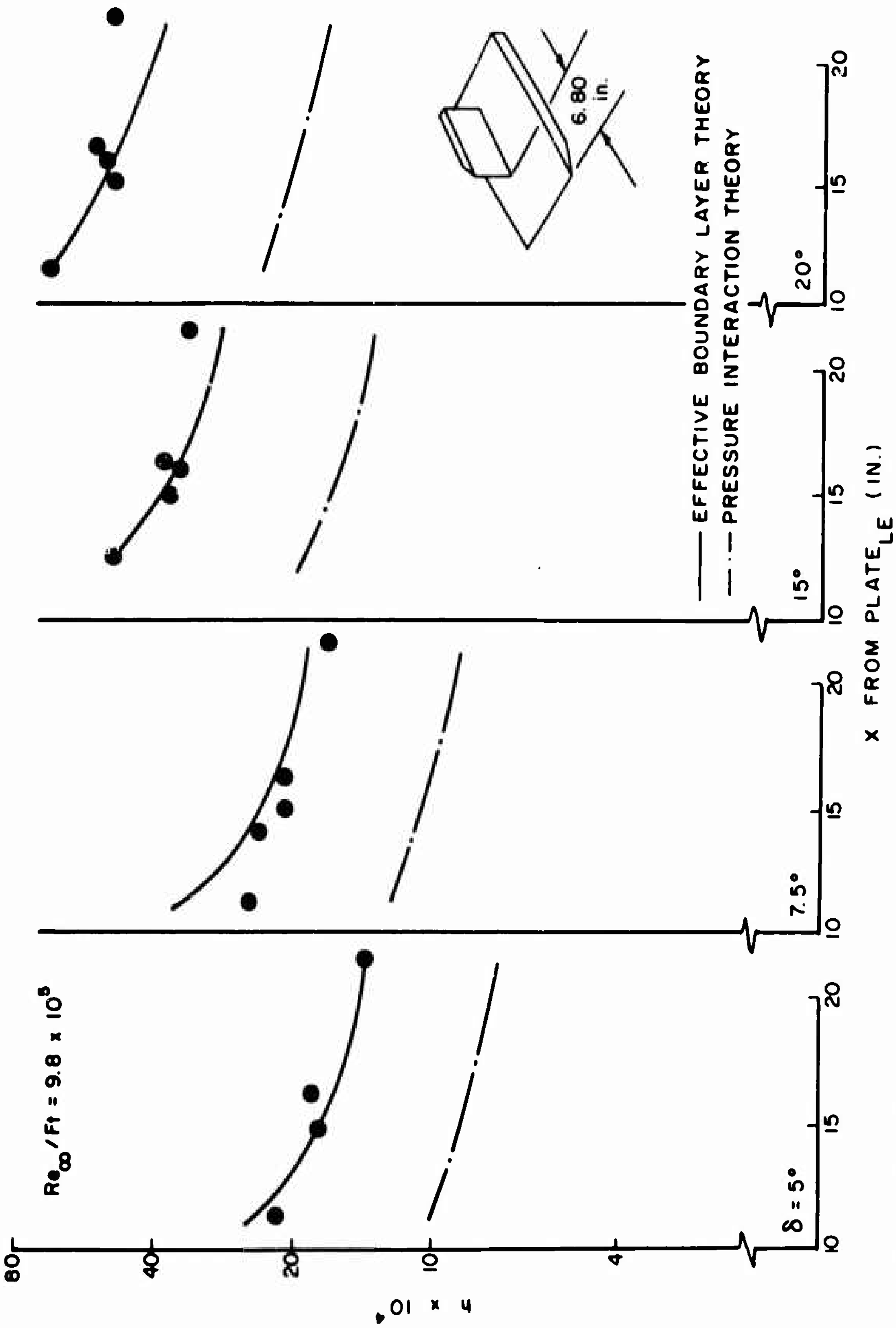


Figure 15. Correlation of the Maximum Heating in the Fin Interaction Region - Mach 8 Laminar Flow

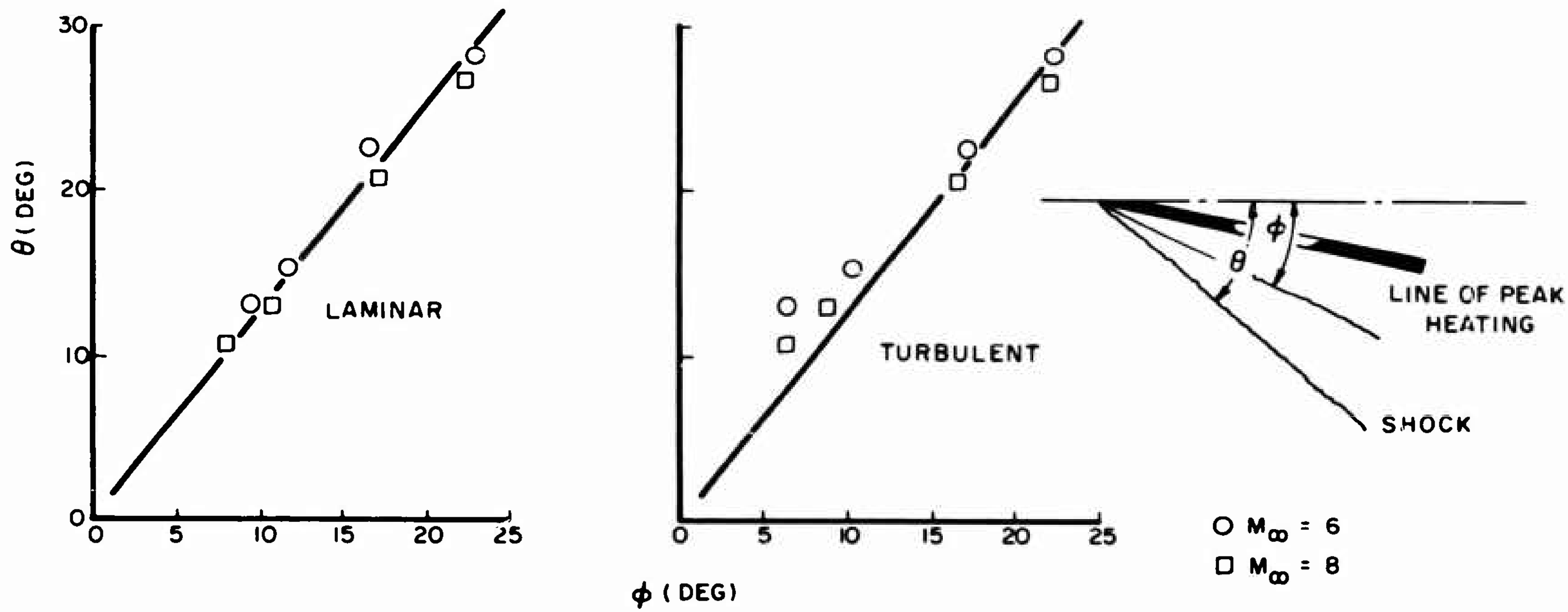


Figure 16. Location of the Ray Angle for Peak Heating Relative to the Inviscid Shock Wave

data did exhibit greater divergence from the correlating line at lower shock angles and Mach numbers. With this relation, a more general expression for the reference length used in the heat transfer coefficient was derived. Appendix III indicates the derivation with the final expression given as $X_{ref} = 0.215\ X_2$, where X_2 is the distance from the fin leading edge to the station under consideration.

With this expression, which is independent to the magnitude of the shock wave angle and using the previously derived pressure correlation for sweep effects, an evaluation of heat transfer data under the influence of swept fins was attempted. Figure 17 indicates the correlation of data at Mach 8 for 5, 7.5, and 15 degrees fin local incidence angles. Sweep angles are varied from zero degree, used as a reference, to 75 degrees. Excellent correlation is noted at 7.5 degrees while at 15 degrees a slight zero shift in the correlation is noted. This zero shift is not a function of sweep but a consequence of the assumptions implicit in the evaluation of the reference length. In spite of this, the correlation indicates a more general applicability of the relationship between peak heating and inviscid shock angle to fins of arbitrary sweep and allows us to circumvent the problem of defining shock angles for swept fins at local angles of incidence. Similar data at Mach 6 are shown in Figure 18. In this figure the solid symbols indicate data suspected of being transitional.

(2) Turbulent Fin Interaction Data

Turbulent heat transfer data were evaluated in a manner similar to that for the laminar data. Figures 19 and 20 indicate correlations of unswept fin data using both the classical pressure interaction theory and the modification to that theory accounting for a new reference length measured from the fin shock wave. Due to the lesser effects of distance on the turbulent heating, the differences between the two theories are not as impressive as in the laminar case. It is apparent from these figures however, that the modified pressure interaction theory forms a conservative upper bound to the data. Distributions of peak heat transfer in the interaction regions at Mach 6 and 8 are compared with data in Figures 21 and 22. Another point of concern was the contrary data trend with distance noted in all the data. While it is not possible to assure the reader of the reasons for this trend, it is plausible and within the framework

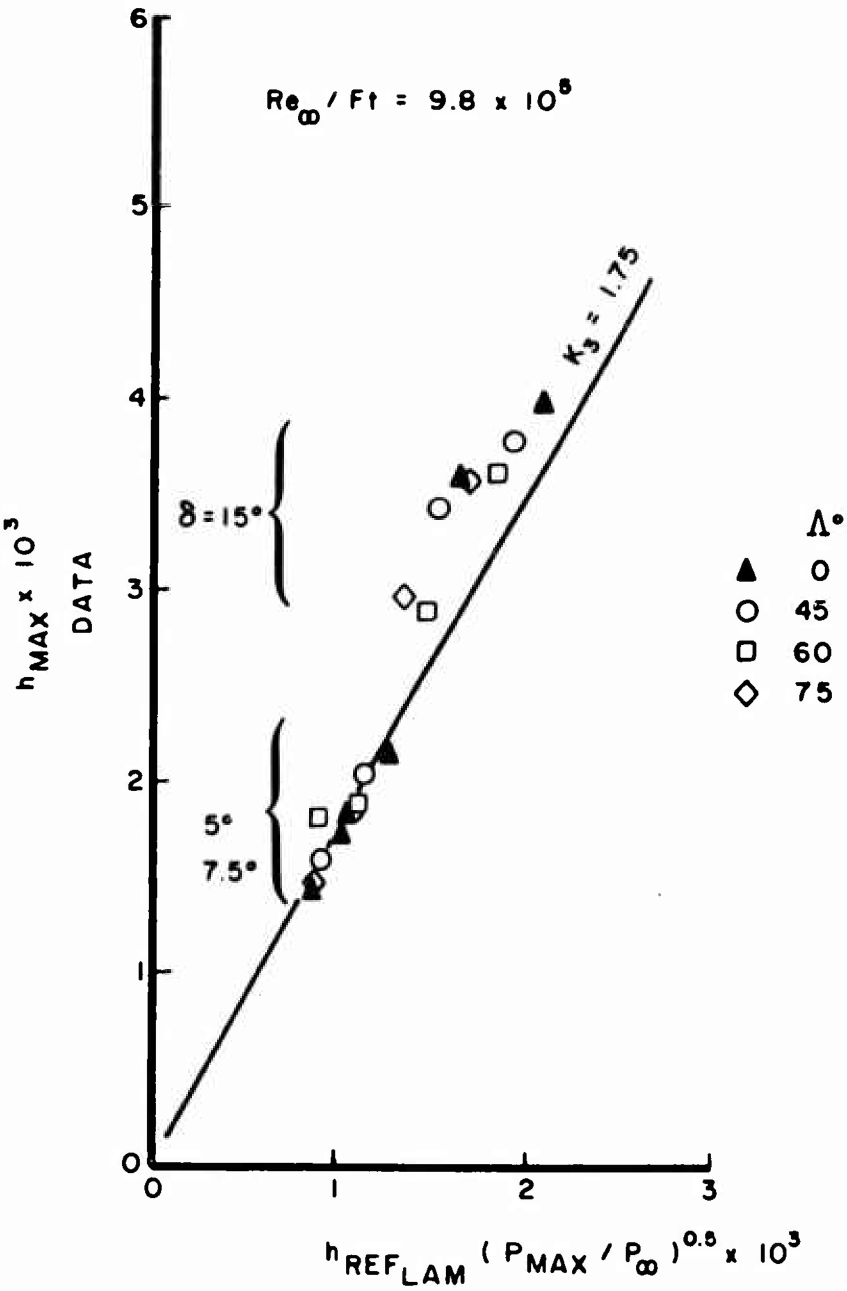


Figure 17. Correlation of Peak Heating Data in the Fin Interaction Region for Fins of Varying Sweep Angle - Mach 8 Laminar Interaction

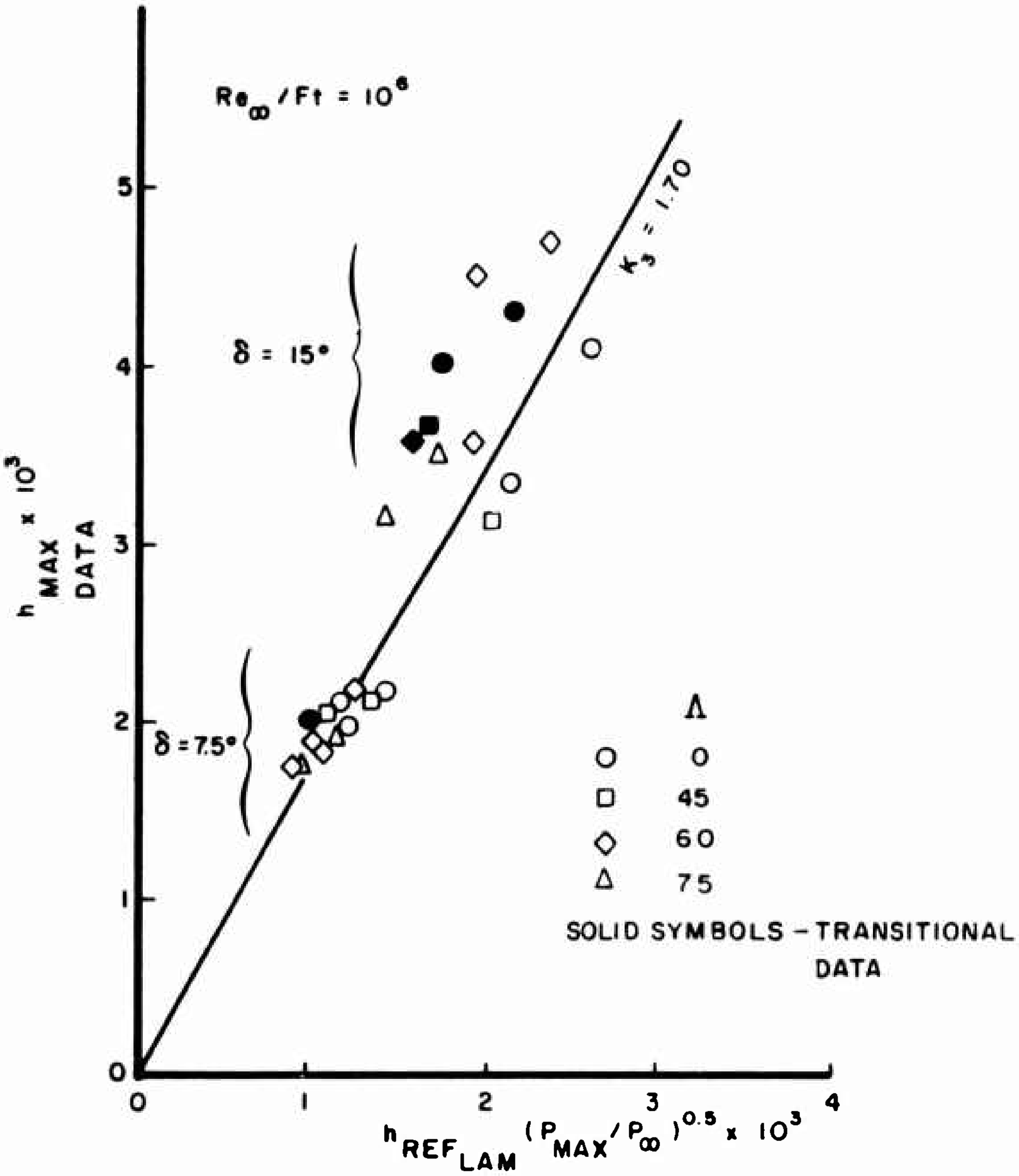
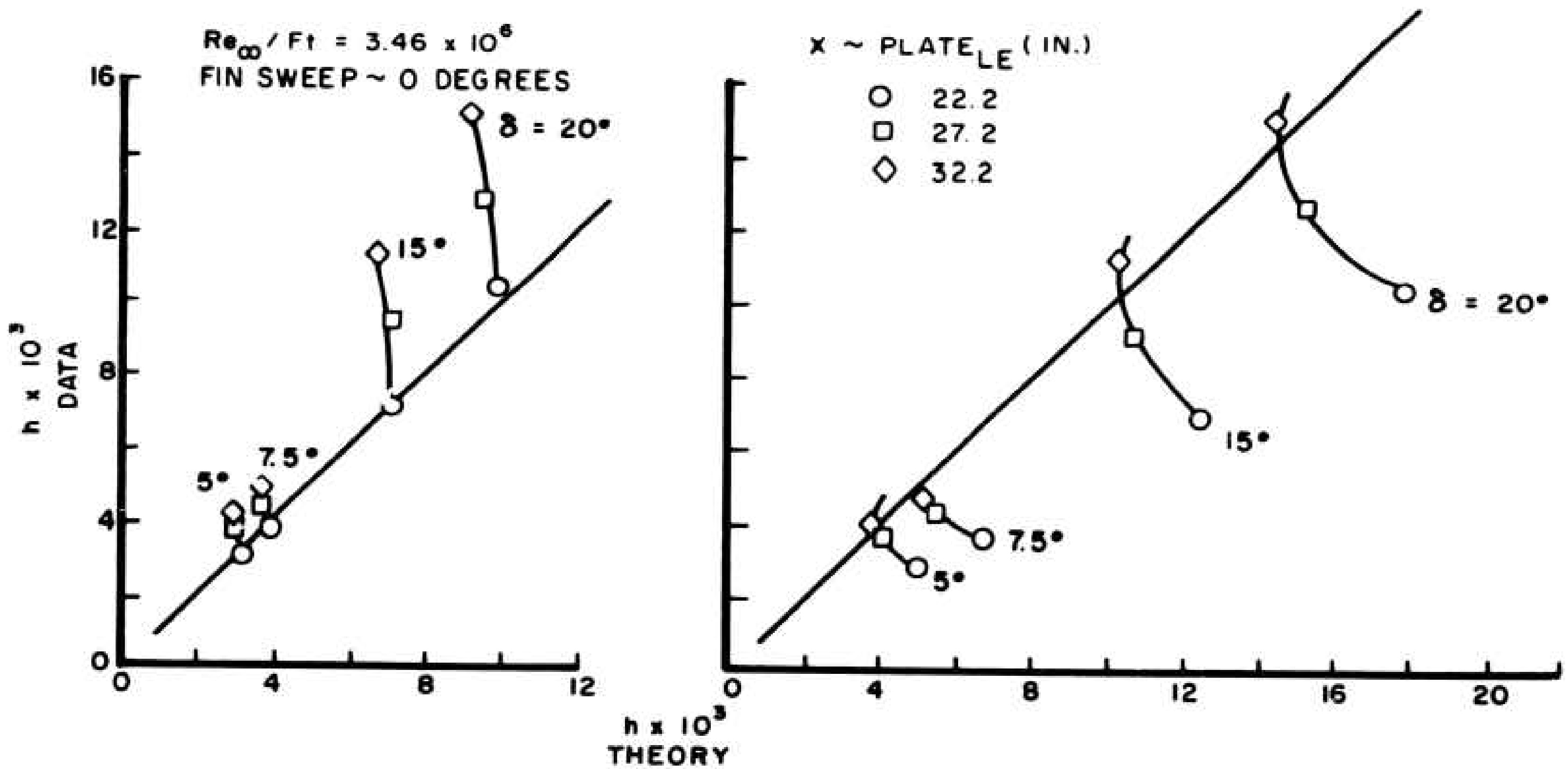


Figure 18. Correlation of Peak Heating Data in the Swept Interaction Region - Mach 6 Laminar Interaction



(A) X DISTANCE MEASURED FROM THE PLATE LEADING EDGE

(B) X DISTANCE MEASURED FROM THE FIN SHOCK WAVE

Figure 19. Correlation of Mach 6 Turbulent Fin Interaction Data With Theory

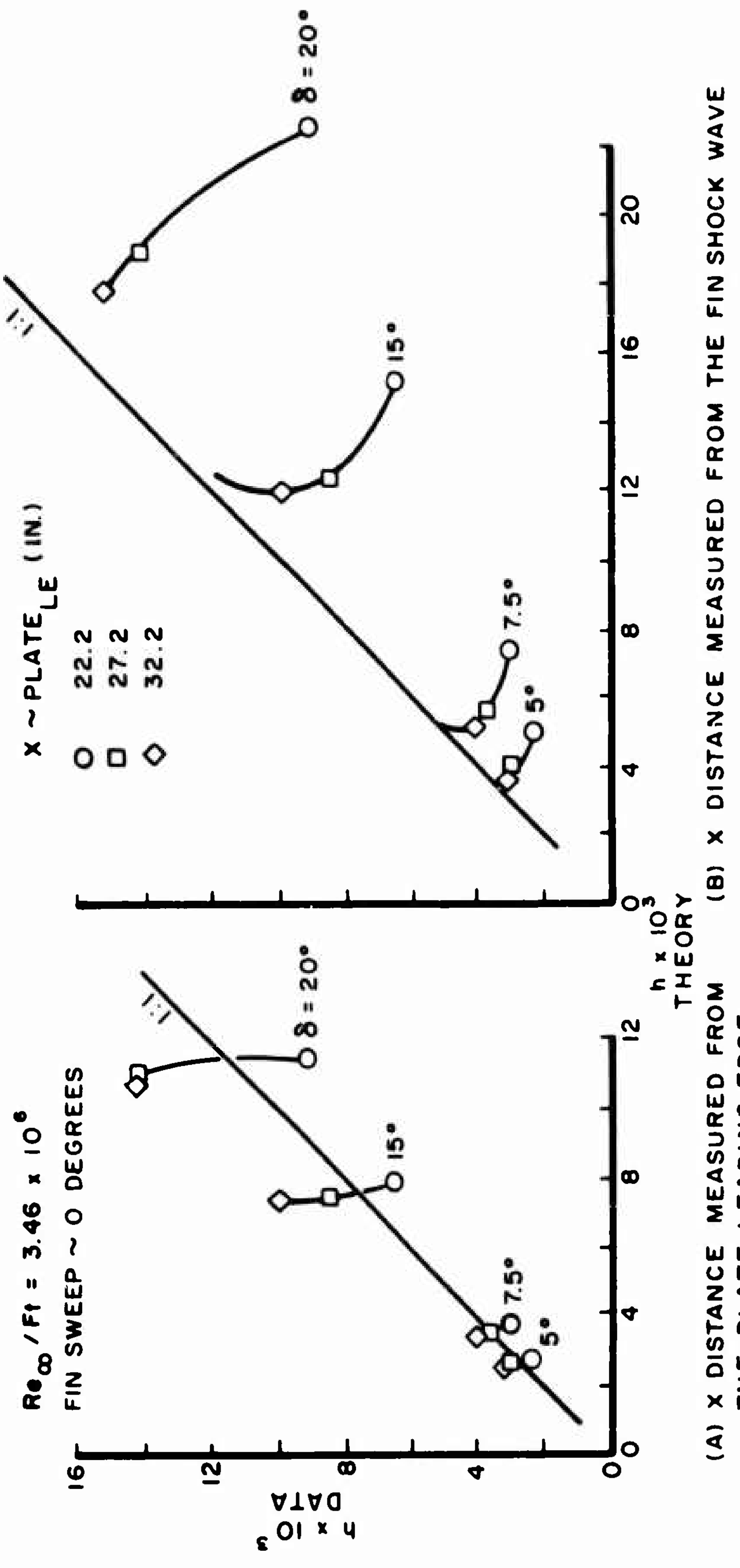


Figure 20. Correlation of Mach 8 Turbulent Fin Interaction Data With Theory

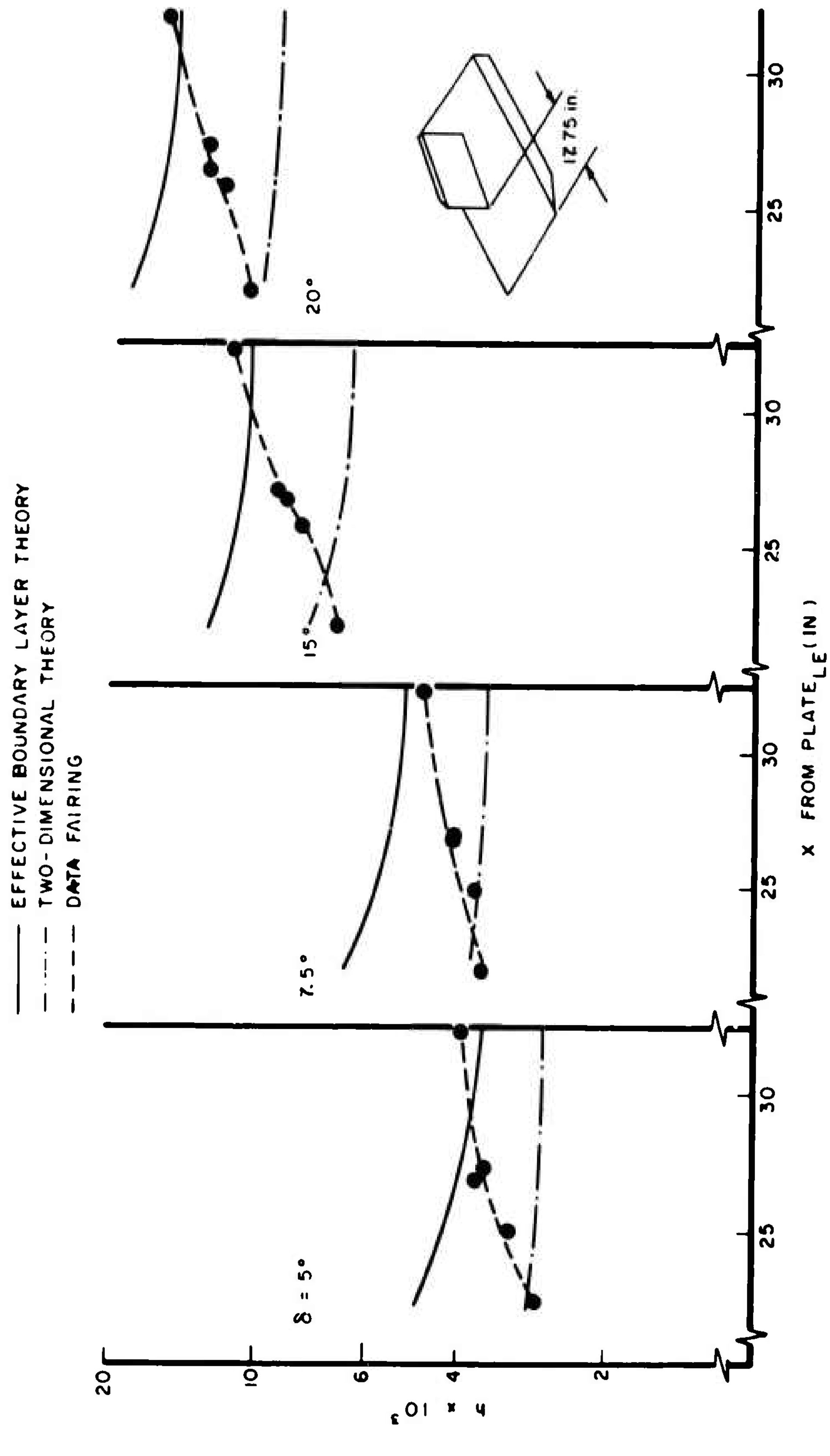


Figure 21. Correlation of the Maximum Heating in the Fin Interaction Region - Mach 6 Turbulent Flow

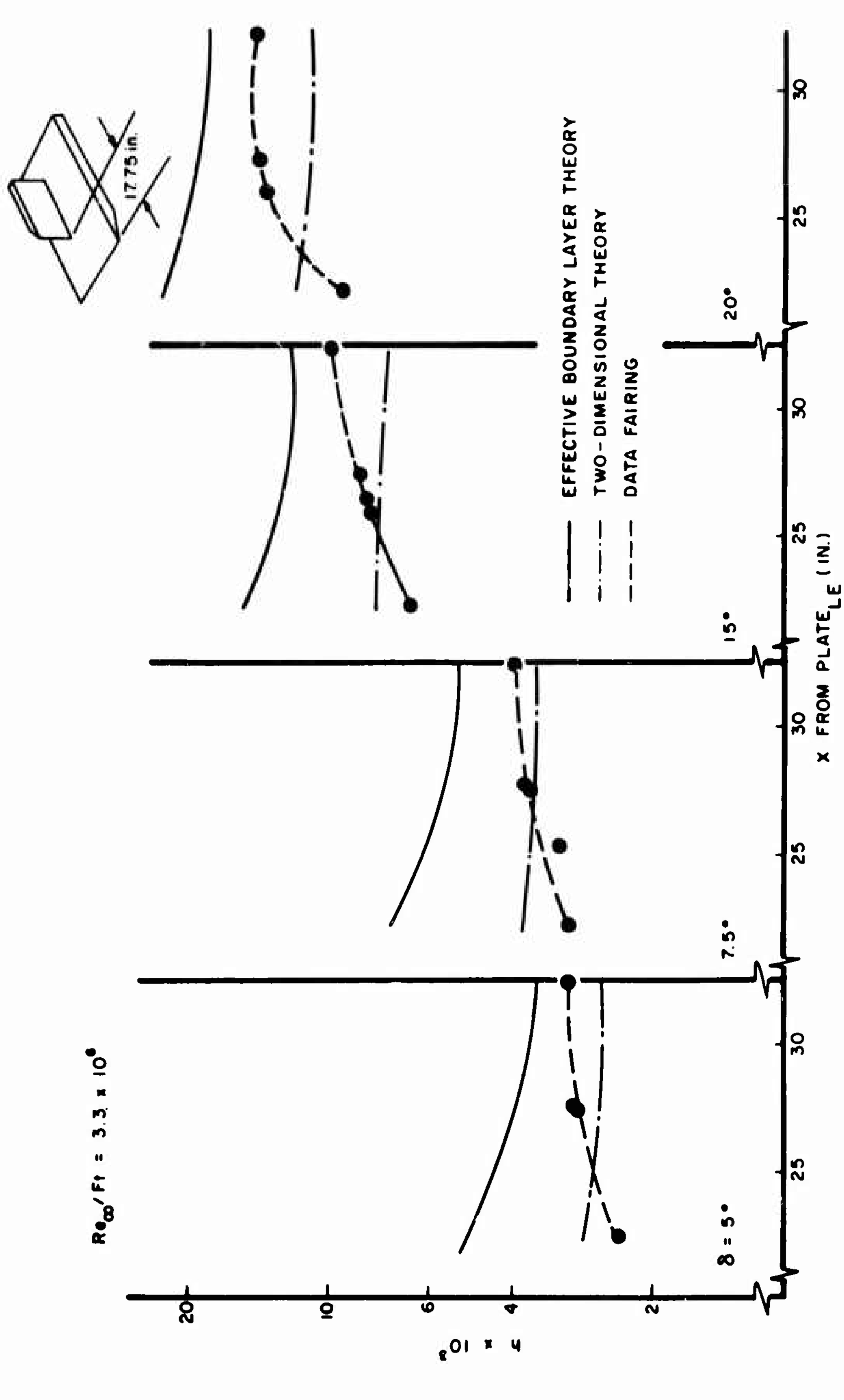


Figure 22. Correlation of the Maximum Heating in the Fin Interaction Region - Mach 8 Turbulent Flow

of our approach to consider this data trend as an indication of boundary layer transition.

Irrespective of the cause, it is important to note that substantial distances or 15 inches or more were required to achieve a stable turbulent interaction. Due to this, no turbulent data on swept fin interactions were correlated nor is such data presented in this report.

SECTION IV

DISCUSSION OF RESULTS

The present test effort and data analysis have contributed to an increasing volume of results in the area of three-dimensional flow field interactions. Unlike the more orderly two-dimensional interaction case previously discussed, the available results on the three-dimensional case pose many unanswered questions. In this section, a review of the work of other authors and a unified discussion of both results and problem areas are made.

Early work on three-dimensional fin interactions was conducted by Miller and coworkers at the Boeing Company largely as an outgrowth of Dyna-Soar efforts. Papers were presented on the evaluation of fin data at both Mach 16 (Reference 3) and Mach 8 (References 5 and 10). The pressure interaction theory approach was first used by Miller in the successful correlation of the Mach 16 data. Most interesting was the drastic change of concept necessitated by the Mach 8 data. In effect, Miller was forced to go to a new reference length to accomplish correlation of the Mach 8 data.

Based upon our success in correlating Mach 6 and 8 data, one must ask why the Mach 16 data correlated with a substantially different application of the theory. While this is an area for more research activity, some insight may be gathered from the fact that substantial distances are required to achieve stable laminar interactions. Figure 23 indicates a composite of data taken in the Mach 8 Tunnel B facility. The solid symbols are the data of Miller presented both in the referenced AEDC data report and in Reference 5, while the open symbols indicate similar data from the present investigation. The data are all referenced to the theory developed in this report. Data from the original work of Miller at Mach 8 were extracted directly from the AEDC output and were not corrected for conduction errors which were estimated to reduce the measured gage output by roughly 30%. It is apparent from this figure that even for the case of the sharp leading edge fin a distance of at least 6 inches is required to achieve a correlation which is independent of distance. It is our view that this distance is Mach number dependent and that it increases with Mach number. If this conjecture proves correct, and it will

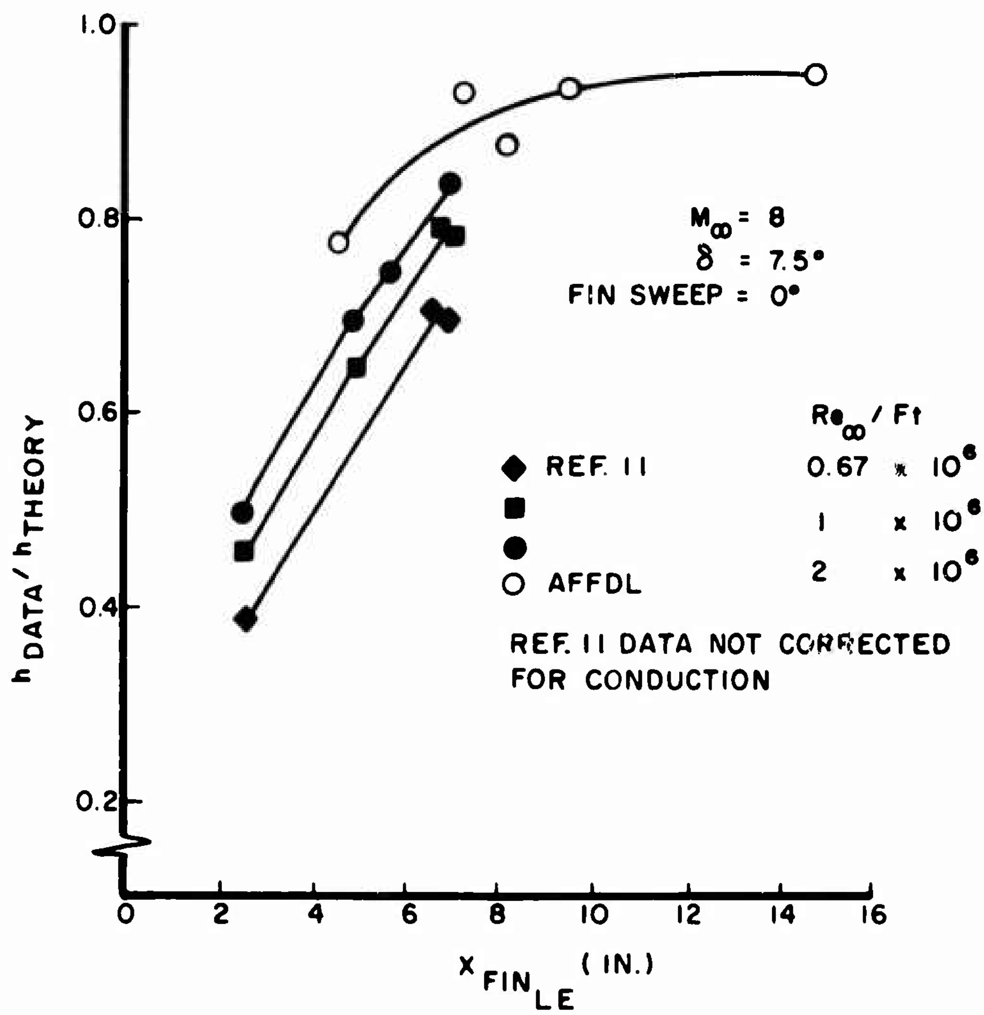


Figure 23. Comparison of Interaction Data Taken for This Report With Previously Generated Data

require additional data to validate, then the Mach 16 data should correlate with our method further downstream.

Another example of this distance to achieve a stable interaction is to be found in a very recent paper presented by Stainback (Reference 4). Operating at conditions near our Tunnel B experiments, Stainback observed a nearly constant variation of peak Stanton number with Reynolds number (distance). The two cases presented in Reference 4 are reproduced as Figure 24 and indicate the value of the theory presented in this report to bound and correlate the interaction data far from the fin leading edge.

The interaction pressure increment used in the theory is yet another area of discussion. Miller in his correlation of the Mach 8 interaction data (Reference 5) used as this increment the difference between the plateau and peak pressure. We have used the difference between the free stream and the peak pressures as the pressure increment. Concurrent evaluation of several test points using both methods indicates to us an improved correlation capability through ignoring the separation. This is most evident when the local incidence angle of the fin is small and the plateau and peak pressures approach eath other. Further, one cannot know a priori for which conditions separation will occur and, in areas of uncertainty as to separation, two values of the estimated peak heating level are possible and equally probable.

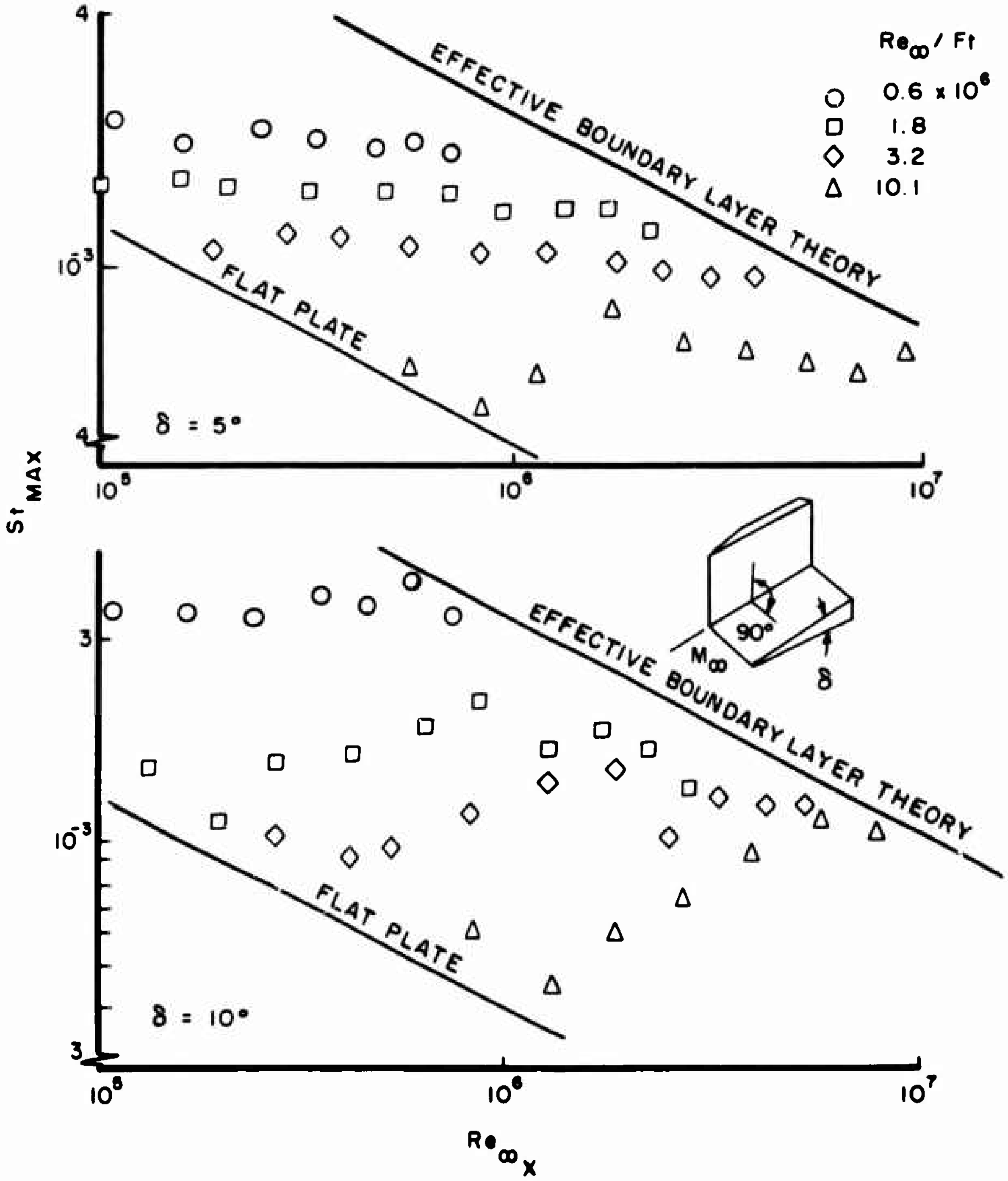


Figure 24. Correlation of Stainback's Mach 8 Fin Interaction Data

SECTION V

CONCLUSIONS

Detailed experimental data have been generated on both two- and three-dimensional interactions in hypersonic perfect gas flows. Pressure and heat transfer data were taken to develop empirical correlations for the location and magnitude of heat transfer due to shock wave impingements which are found on typical airbreathing cruise systems.

Two-dimensional interaction data agree well with the classical pressure interaction theory and for design applications the problem reduces to the estimation of impingement pressures for which both hand calculations and computer programs have been developed.

It has been demonstrated that three-dimensional interactions disrupt the basic plate boundary layer to the extent that a new effective boundary layer is initiated. In both laminar and turbulent flow the method developed in this report and based on this premise correlates the data after a finite distance required to stabilize the perturbed boundary layer. In the turbulent case this distance is extremely long and is thought to be caused by transition of the new fin-induced boundary layer which is initially laminar.

The distance required to achieve a stable and predictable interaction heating pattern is a point of concern in the subscale testing of large hypersonic cruise configurations and will require both additional experimentation as well as some scaling laws to employ properly the results of ground tests.

While the data shown in this report were for the sharp fin case, the method is equally applicable to the blunted fin case. Chief difficulties in its application to the blunt fin are the more complex shock shape of a blunt swept fin at angle of attack and the more involved relationship between the inviscid shock and location of peak heating. In general, the sharp fin case presents the more severe heating problem due to the closer proximity of the shock to the body. Insufficient data were generated to allow a criterion to be established on the

zones of application for laminar and turbulent analysis. Due to the severity of the heating, such a criterion should be established to preclude overly conservative design.

Fin leading edge sweep angle has been shown to reduce the heating only slightly. The pressure decay with sweep was found to vary only as the cosine to the 0.3 power. Correlation of swept fin data with the simplified concepts of this report have been demonstrated.

APPENDIX I

LAMINAR-TURBULENT HEATING RELATIONSHIP

A generalized expression for flat plate heating in a perfect gas is rewritten from Reference 12 as

$$St\, Re_{\infty_X}{}^{n} = A\left[\frac{U_e}{U_\infty}\frac{P_e}{P_\infty}\right]^{1-n}\left[\frac{T_\infty}{T^*}\right]^{1-2n} C^{*n}$$

where

A = 0.332 and n = 0.5 for laminar flow

A = 0.0296 and n = 0.2 for turbulent flow

The ratio of turbulent to laminar reference heat transfer coefficients at the same conditions becomes

$$\frac{h_{TURB}}{h_{LAM}}\;\frac{Re_{\infty_X}{}^{0.2}}{Re_{\infty_X}{}^{0.5}} = \frac{0.0296}{0.332}\;\frac{\left(\frac{U_e}{U_\infty}\frac{P_e}{P_\infty}\right)^{0.8}\left(\frac{T_\infty}{T^*}\right)^{0.6} C^{*0.2}}{\left(\frac{U_e P_e}{U_\infty P_\infty}\right)^{0.5} C^{*0.5}}$$

$$\frac{h_{TURB}}{h_{LAM}} = 0.0892\; Re_{\infty_X}{}^{0.3}\left(\frac{T^*}{T_\infty}\right)^{-0.6}$$

where only the dependence on Reynolds number and reference temperature has been retained as significant.

APPENDIX II

PRESSURE INTERACTION THEORY

The generalized expression for flat plate heating used previously is

$$St\, Re_{\infty_x}^{\,n} = A \left[\frac{U_e\, \rho_e}{U_\infty\, \rho_\infty}\right]^{1-n} \left[\frac{T_\infty}{T^*}\right]^{1-2n} C^{*n}$$

where

A = 0.332 and n = 0.5 for laminar flow

A = 0.0296 and n = 0.2 for turbulent flow

For the same free stream conditions at a given location on the plate and for an arbitrarily imposed pressure ratio

$$\frac{h_{P_{MAX}}}{h_{REF}} = \left[\frac{\left(\frac{U_e}{U_\infty}\right)_{P_{MAX}} \left(\frac{\rho_e}{\rho_\infty}\right)_{P_{MAX}}}{\left(\frac{U_e}{\rho_\infty}\right)_{P_{REF}} \left(\frac{\rho_e}{\rho_\infty}\right)_{P_{REF}}}\right]^{1-n} \left[\frac{\left(\frac{T_\infty}{T^*}\right)_{P_{MAX}}}{\left(\frac{T_\infty}{T^*}\right)_{P_{REF}}}\right]^{1-2n} \left[\frac{C^*_{P_{MAX}}}{C^*_{P_{REF}}}\right]^{n}$$

It is further assumed that

(1) the velocity, U_e, does not vary significantly from free stream values in the higher pressure interaction region so that

$$\left(\frac{U_e}{U_\infty}\right)_{P_{MAX}} \approx \left(\frac{U_e}{U_\infty}\right)_{P_{REF}}$$

(2) that a linear temperature viscosity relation exists so that

$$(C^*)_{P_{MAX}} \approx (C^*)_{P_{REF}}$$

(3) and that the reference temperature is not affected by the interaction so that

$$\left(\frac{T_\infty}{T^*}\right)_{P_{MAX}} \approx \left(\frac{T_\infty}{T^*}\right)_{P_{REF}}$$

The final approximation becomes

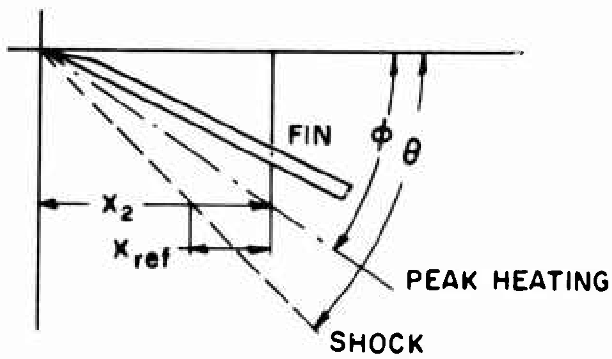
$$\frac{h_{P_{MAX}}}{h_{P_{REF}}} \approx \left(\frac{P_{MAX}}{P_\infty}\right)^{1-n}$$

and this, used by Miller, Sayano and others, is referred to in this report as the pressure interaction theory.

The collective error of the above three assumptions was evaluated by expressing each as a function of the pressu [illegible] se. It was found that use of the approximate expression is conserva [illegible] hat the assumptions are more restrictive in a turbulent than in a lam [illegible] ary layer.

APPENDIX III

RELATION BETWEEN SHOCK WAVE AND LINE OF MAXIMUM HEATING



Equating the normal distances between the center line and peak heating point yield

$$(X_2 - X_{ref}) \tan\theta = X_2 \tan\phi$$

It was found empirically (Figure 16) that $\phi = 0.785\theta$. Thus, with the approximation that $\tan\theta = \theta$ and $\tan\phi = \phi$, then

$$X_{ref} = 0.215\, X_2$$

REFERENCES

1. Sayano, S. "Heat Transfer in Shock Wave-Turbulent Boundary Layer Interaction Regions", Douglas Report SM 42567, November 1962.

2. Fabish, T. J. and Levin V., "Thermal Effects of Shock Wave-Turbulent Boundary Layer Interaction at Mach Numbers 3 and 5," North American Aviation, Inc. NA 62H-795, November 1962.

3. Miller, D. S., Hijman, R., Redeker, E., Janssen, W. C. and Mullen, C. R., "A Study of Shock Impingement on Boundary Layers at Mach 16", Proceedings of the 1962 Heat Transfer and Fluid Mechanics Institute, Stanford University Press, 1962.

4. Stainback, P. C., "Corner Flow at Hypersonic Speeds," Presented at the NASA Conference on Hypersonic Aircraft Technology, May 1967.

5. Miller, D. S. and Redeker, E., "Three Dimensional Corner Flow Separation," Presented at the 48th Meeting of the Bumblebee Panel, 10-11 September 1963.

6. Holden, M., "Separated Flow Studies at Hypersonic Speeds, Part II, Two-Dimensional Wedge Separated Flow Studies," Cornell Aeronautical Laboratory, Inc. Report No. AF-1285-A-13(2), December 1964, AD 613 463.

7. Kutschenreuter, P. H. Jr., et al., Investigation of Hypersonic Inlet Shock Wave-Boundary Layer Interaction, Part II: Continuous Flow Test and Analyses, AFFDL TR 65-36, AD 636 981, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio.

8. Bertram, M. H. and Feller, W. V., "A Simple Method for Determining Heat Transfer, Skin Friction, and Boundary Layer Thickness for Hypersonic Boundary Layer Flows in a Pressure Gradient," NASA Memo 5-24-59L, June 1959.

9. Hankey, Wilbur L., Supervisory Research Aerospace Engineer, Aerospace Research Laboratories, Wright-Patterson Air Force Base, Ohio (Private Communication).

10. Gulbran, C. E., Redeker, E., Miller, D. S. and Strack, S. L., Heating in Regions of Interfering Flow Fields, Part 1, Two and Three Dimensional Interactions at Mach 8, AFFDL TR 65-49, Part 1, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, July 1965.

11. Myers, J. R. and Roberts, B. W., An Investigation of Shock Impingement on Boundary Layers for Wing-Body Configuration at Mach 8, AEDC-TDR-63-222, Arnold Engineering Development Center, Arnold Air Force Station, Tennessee, AD 440 060, October 1963.

12. Wallace, J. E. and McLaughlin, E. J., Experimental Investigations of Hypersonic, Turbulent Flow and Laminar, Leeward-Side Flow on Flat Plates, AFFDL-TR-66-63, Volume 1, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, July 1966.

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13. ABSTRACT

The design of aircraft for sustained operation at hypersonic speeds requires the understanding of aerodynamic heating generated through interfering flow fields. Such interactions not only determine the required level of vehicle thermal protection but also create severe gradients of temperature along skin panels. The Air Force Flight Dynamics Laboratory has completed an extensive experimental program supporting the conceptual design of these vehicles. Experimental results have been generated on models illuminating the basic features of both two- and three-dimensional interactions with results applicable to the design of hypersonic aircraft. This report presents these data and correlations with theory in the Mach number range 6 through 10. Results indicate the applicability of current design practices, areas requiring further investigation, and the problems involved in interpretation and application of interference data from hypersonic facilities to the desired free flight condition.

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